PROPULSION FOR LUNAR DESCENT AND ASCENT

BY

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qualitative or descriptive in nature although quantitative comparisons will be made of various alternatives when appropriate to the presentation. Except for a general picture of the overall manned mission, this paper will be limited to an examination of propulsion to be employed within the moon's sphere of activity; that is, where the moon's gravitational field is more important than the Earth's. To confine the discussion within this restraint, we must assume spacecraft insertion into trans-lunar trajectory by propulsion systems which will not be considered within the scope of this paper.

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INTRODUCTION

Man has ventured to set foot on the moon. In May of 1961 the late President Kennedy announced that the United States would undertake to land men on the moon and return them safely to the Earth's surface. As a preliminary adjunct to the manned flight, several unmanned lunar landings are planned. These efforts have now been underway for several years and most of the basic technical decisions regarding the method of accomplishing the missions have been made.

The purpose of this paper is to examine several aspects of the propulsion problem for performing the lunar landing maneuver and for ascent from the moon's surface. The paper will cover first, the performance requirements of the propulsion equipment in terms of the several path alternatives; second, the arguments for the choice of the lunar rendezvous mode of approach, which has been selected as the method to be used for the manned landing mission; and third, a discussion of the spacecraft propulsion equipment which is under development to perform lunar landing missions. The discussion will be principally

FOREWARD

Before beginning the presentation of the paper prepared for this meeting, I should like to make a few remarks about the general subject. I consider it a privilege to be able to discuss, at this meeting of the XVth International Astronautical Congress, some of the propulsion problems of lunar descent and ascent. Exchanges of views on these very difficult space maneuvers are welcome. Exchanges of viewpoints fluorish in an environment not only of professional respect but also of mutual benefit. The United States of America stands ready to join in cooperative space ventures with any nation. Its program is an open program and the benefits resulting from it are shared with all. In this spirit I shall review the calculations, the decisions, and the propulsion equipment for the lunar landing mission.

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PROPULSION REQUIREMENTS FOR LUNAR MANEUVERS

The propulsion requirements for performing a lunar landing mission are characterized by the velocity change (ΔV) necessary to bring a payload to rest at the lunar surface. The ideal velocity change requirement is, of course, dependent on the trajectory flown from the Earth, the relative phase-relationship of the earth and the moon, the time of launch, the planned duration of flight, the longitude and latitude of launch, and the longitude and latitude of landing. It is not within the scope of this paper to examine these variables in detail; this analysis has been covered in other papers.

To the ideal trajectory velocity change requirement must be added reserve requirements imposed by flight path corrections due to accelemeter errors, altitude determination errors, impulse errors, and possible pilot errors.

Of the many variables that affect the velocity change requirement the duration of flight exerts the greatest influence. Since the purpose of the propulsion system is to propel a useful payload to its destination and it is in the mission interest to maximize that payload, the most attractive flight paths are those with flight durations above 70 hours where ΔV requirements approach a minimum. The decision to accept such flight duration then establishes boundaries for thrust-to-spacecraft-mass ratio, impulse accuracy and, for the landing phase, throttling range, which we will now examine.

There are two general classes of lunar landing methods. The first of these is descent directly from a lunar trajectory. The second is more complex, entailing first an entry into a lunar orbit and then a descent to the surface. In addition, propulsion equipment to perform trajectory corrections during the earth-moon or moon-earth legs of the mission is also required. These maneuvers generally lie outside the moon's sphere of activity and their requirements will not be discussed in this paper. In addition for manned missions, which necessitate return, either path entails hovering over the lunar surface prior to final let-down to permit the astronauts an opportunity to study the surface and select a site for landing. If only part of the total expeditionary spacecraft system descends, leaving a parent ship in the lunar orbit, the additional maneuver of lunar orbit rendezvous between the two spacecraft must also be performed.

The requirement of each of these maneuvers needs to be examined, along with the advantages or disadvantages of launch opportunities or of abort situations.

Descent and Landing

For this discussion the landing maneuver will be considered as a descent which reduces the spacecraft velocity to zero at a low altitude above the moon's surface followed by a landing maneuver from a low hover altitude to a landing on the lunar surface.

Descent Directly from Trans-lunar Trajectory

The "ideal" mode of landing on the moon is descent directly to the surface from a trans-lunar trajectory. The approach velocity of the spacecraft relative to the moon is dependent on the flight duration, as well as on other factors already mentioned. Figure 1 relates the transit time to the hyperbolic excess velocity relative to the moon (ΔV_∞) and also presents the minimum total impulse velocity increment (ΔV_o) requires for a direct landing both for times of maximum separation of the moon and the Earth (1.33×10^9 feet) and minimum separation (1.17×10^9 feet). The hyperbolic excess velocity is a measure of the trajectory energy and would be the velocity of the spacecraft relative to the moon at' impact if the moon had no gravity.

The minimum total impulse velocity increment represents a hypothetical "ideal" landing where the propulsive impulse is applied in infinitesimal time and is called the impulsive velocity increment requirement.

For direct landing with finite thrust, which is substantially more realistic, the total impulse required is dependent on the spacecraft velocity vector relative to the surface, as well as the time taken to execute the maneuver; vertical descent requires the maximum velocity increment. The characteristic velocity differs from the impulsive velocity requirement in that it includes the added propulsive requirements necessary to overcome the gravitational attraction vector during the flight period; it is equivalent to the velocity attainable by the same propulsion system operating for the same time in linear flight in a gravitationless vacuum. Figure 2 shows this vertical characteristic velocity change requirement as a function of the thrust-to-initial-mass ratio.

The landing is governed by the same basic equations that govern ascent (with appropriate mathematical sign, of course) and those familiar with ascent trajectories can readily deduce the effects of other-than-vertical flight paths.

Note in Figure 3 that characteristic velocity penalties are quite significant unless wide range variable thrust is available. This result suggests that for direct descent to landing, the use of two propulsive systems may be advantageous; one for deceleration, the other to complete the landing.

Figure 3 illustrates the effect of thrust-to-initial-mass ratio on ignition altitude for a vertical descent. Since errors in altitude measurement at the beginning of propulsive thrust are reflected in errors in altitude at the burn-out point, burn-out errors in altitude are likely to be greater when low thrust-to-mass-ratio systems are used. Quite obviously large uncertainties in propellant loading at this point in the flight could have serious effects.

The descent touch-down on the lunar surface is probably the most critical period of flight during the entire lunar mission. Accordingly, the possibility of any equipment failure or an unsuitable surface condition is an extremely important consideration. Our first thought would be to use the propulsion stage designed for lunar launch to perform the abort maneuver.

At some points in the direct descent from lunar trajectory, these abort requirements are in excess of the normal requirement for launch from the surface. If the trans-lunar trajectory path makes an angle less than 45 degrees with the lunar surface then the spacecraft can be turned from its path (provided that the ascent stage has sufficient thrust) and put into an elliptical orbit around the moon from which it can be returned to Earth without exceeding the characteristic velocity requirement for launch from the moon to Earth. Figure 4 illustrates the problem.

Since landing sites of greatest interest are on the visible portion of the moon and since landing on these sites generally requires approach paths greater than 45 degrees with respect to the surface, it follows that the direct landing trajectory does not appear attractive for manned missions. It is attractive for unmanned missions, where no abort requirements need be considered, because only one continuous propulsive burning time is required, and line-of-sight communication is possible.

Descent via Lunar Orbit

Deceleration to a lunar orbit permits survey of the lunar surface around the belt of the orbit prior to commitment to land. Since much information about the moon's gross surface characteristics can be obtained from a relatively low orbit and since, as we shall see, abort problems are greatly alleviated during the descent maneuver, the use of the lunar orbit is very appealing for manned flight in spite of the additional total characteristic velocity requirements as compared to direct descent.

Figure 5 illustrates the total impulsive velocity increments for descent by use of a lunar circular orbit as a function of the hyperbolic excess velocity relative to the moon. For low altitudes (50 to 100 miles) the additional velocity increment is perceived to be of the order of one hundred feet per second. These figures do not include any velocity requirement to effect an orbit plane change.

Figure 6 shows the impulsive velocity requirements for entry into various circular lunar orbits as a function of the hyperbolic excess velocity relative to the moon. These curves also apply to exit from the lunar orbit to return to Earth.

The effect of thrust-to-initial-mass on the characteristic velocity requirement for one particular approach speed is shown on Figure 7. For this maneuver thrust-to-mass ratio effect is relatively small; even for values of 0.1, the additional velocity requirement is less than 100 feet per second. The effect of specific impulse on the characteristic velocity requirement is also quite small. Thus, entry into orbit maneuvers can be accomplished with relatively low thrust propulsion units. These curves are very nearly correct for exit from the lunar orbit to return to Earth.

Descents from lunar orbit to the hover altitude are attractive, in comparison to direct descents, because propulsion system of relatively low thrust-to-mass ratios and throttling ratios can be used without severe penalty. Landing from orbit also allows abort within the capability of the ascent stage at all times during descent. Figure 8 illustrates the abort maneuver. Figure 9 indicates typical characteristic velocity requirements for abort during descent from a particular orbit.

While the direct lunar landing is more efficient than landings employing a lunar orbit for the idealized impulsive thrust case, the advantages of the direct flight path are largely lost when finite thrust propulsion systems are considered. The additional system weight required to meet the higher thrust-to-mass ratios of the direct landing offsets the slight velocity increment advantages. In fact, for thrust-to-initial-mass ratios below approximately 4.0 the velocity increment advantage of the direct descent as compared to the lunar orbit and Hohman transfer descent has vanished.

There are five principal approach trajectories to be considered for descent from lunar orbit. Four are illustrated in Figure 10a & 10b. Each has certain advantages and disadvantages. They are:

1) Continuous Constant-Thrust Descent: The descent propulsion system is ignited in orbit and the descent to hover altitude made at constant thrust following an optimum flight path. There is no interruption of propulsive burning. This method requires precise path and position measuring equipment along with an exact knowledge of propellant loading, as does the next method.

2) Continuous Variable-Thrust Descent: Ignition occurs in orbit with descent at minimum thrust along an optimum path; at a particular point during flight the throttle is advanced to nearly full thrust for the remainder of the descent to hover altitude. By using the variable thrust capability the astronaut can make adjustments near the end of descent so that the spacecraft achieves the hovering altitude. This final approach technique is probably required if the spacecraft is to be landed at a particular location on the surface.

3) Hohman Ellipse Transfer and Descent: The descent propulsion systems must ignite briefly at the point of the lunar orbit (assumed to be circular) just opposite the landing point. As the spacecraft nears the landing site the descent propulsion system must be restarted and descent made to hover altitude following an optimum flight path. Although this path is most efficient in terms of characteristic velocity requirement, it is obvious that the landing site is not in view during the early part of the descent and that the propulsion system must be operated at least twice.

4) Synchronous Ellipse Descent: The descent propulsion system first transfers the spacecraft from its initial circular orbit to an elliptical orbit of equal period but with a low-altitude pericynthion near the landing site. At the pericynthion the descent propulsion system is restarted and an optimum-path descent made to hover altitude. The ignition point for the first maneuver can be either 90 degrees or 270 degrees from the landing site. This maneuver is attractive if a small landing spacecraft is disjoined from a parent spacecraft which remains in circular orbit. The maneuver permits both spacecraft to remain in sight of each other through several inspection passes and during the descent.

5) The Near-Vertical Descent: It is also possible to use a high-thrust, two-burn descent which requires re-ignition of the propulsion system to prevent impact with the lunar surface. Such trajectories can reduce the angular range between the initial firing and the landing to 25 degrees without significantly increasing the characteristic velocity increment requirement. Propulsion system failure at the second ignition, however, would leave the spacecraft on an impact trajectory. Such descents are not favored for manned missions.

Figure 11 shows the relationship between the minimum characteristic velocity increment and system maximum-thrust-to-initial-weight ratio for four of these landing approach paths. The abscissa can be transformed into a throttling ratio if the spacecraft is assumed to have a final ratio of lunar-weight-to-engine-thrust and this scale has been added for the value of spacecraft minimum thrust to lunar landing weight of 0.75. (This violates the concept of constant-thrust burning; however, it is obvious that variable thrust would be required for descent from the hover altitude.)

The Hohman ellipse descent requires the least velocity increment at all throttling ratios; the constant-thrust descent the most. The synchronous descent maneuver requires almost 300 feet per second more velocity change than a Hohman transfer; the variable-thrust descent lies between the Hohman ellipse transfer requirement and the synchronous-orbit requirement. Except for the constant-thrust method, which goes out of sight for orbits of 100 miles, the velocity differences in these paths are quite independent of spacecraft propulsion throttling ratios.

When practical values of engine thrust chamber weights are plugged into these relationships, it becomes clear that beyond maximum-thrust-to-initial-mass ratios of 1/3 (throttling ratio about 5.5) the spacecraft propulsion system weight, including propellants, remains essentially constant. The differences between the most efficient path (Hohman) and the least efficient path (synchronous orbit) amount to about two percent of total spacecraft weight. Using a single engine with a throttling ratio of 3.5:1 results in a penalty of about three percent of total spacecraft weight with respect to the best throttling ratio. Since a throttling range of 3.5:1 is possible in a fixed-orifice engine injector system, it is clear that use of such a thruster configuration would not be prohibitive in total spacecraft system weight. It would, however, limit capability for abort or other emergency-demand maneuvers.

Hover and Landing

Hovering at altitude prior to continuing to the surface increases the total characteristic velocity increment over that for an "ideal" descent without hover. Figure 12 shows the increase in total characteristic velocity requirement as the hover altitude is increased.

For manned landing the propulsion system must provide some capability to remain aloft for a limited time to permit the astronauts to study the surface and select an appropriate site for landing. To do this requires continuous thrust equal to the spacecraft lunar weight (the mass of the spacecraft X the lunar gravitational acceleration, which is approximately 5.31 feet/second/second). During this time limited translation horizontally to the surface is also possible. The characteristic velocity requirements for simply remaining aloft are the product of the lunar gravitational acceleration and the time of burning.

Now, having arrived at zero velocity with respect to the moon at an (assumed) respectful distance from the surface we perceive that to land vertically we must reduce the spacecraft thrust to some value below the spacecraft's lunar weight. For this touch-down maneuver a thrust level of 3/4 of the spacecraft weight is representative of the minimum thrust required to perform the landing efficiently. Figure 13 shows more exactly the characteristic velocity requirement to descend from a hovering altitude of 1,000 feet as a function of the thrust-to-lunar-weight ratio with throttling (maximum thrust to minimum thrust) ratio as a parameter. Figure 14 illustrates the variation of characteristic velocity increment with variation in throttling ratio for descents from several hovering altitudes. These figures show that if hovering can be performed at the lower altitudes, relatively small throttling ratios can be employed with only modest and characteristic velocity penalties. If high hovering altitudes are required, then either multiple engines, or one engine with a wide throttling capability must be provided.

An alternate scheme of operation is to provide one engine with high-response start and stop characteristics. The requirements of this scheme of operation are not illustrated by the figures.

In actual flight a true hover point (zero vertical velocity) may not be used, particularly if a wide-range throttling engine is available to control the spacecraft velocity as it approaches the surface.

Ascent from Lunar Surface

Direct Return to Earth

The propulsive requirements for direct ascent to a trans-Earth trajectory are similar but less stringent than the landing maneuver. Obviously variable thrust is not required in ascent. The characteristic velocity requirement for various thrust-to-initial mass ratios for vertical flight (maximum velocity increment requirement) are essentially those shown in Figure 2 for descent. Ascents at an angle to lunar surface with paths following the curvature of the moon minimize the lunar gravitational loss term in the ascent equations, in the same manner as for Earth ascents.

Ascent to Lunar Orbit

Launch to lunar orbit prior to return to Earth will be required for many of the landing sites on the moon; it may be desirable to extend the launch "window" for return to Earth in any mission; for lunar orbit rendezvous with a parent spacecraft a launch to orbit is mandatory.

Three types of ascent trajectories are shown in Figure 15 and their characteristic velocity increment requirements are shown in Figure 16 in terms of take-off thrust-to-mass ratio. The variable-thrust ascent is not shown because variable thrust seems an unnecessary and unwarranted complication for the ascent maneuver. For the two-burn elliptical transfer ascent paths, a pericynthion altitude of 50,000 feet was assumed. These transfer ellipse curves show a minimum characteristic velocity requirement at a thrust-to-initial-mass ratio of 0.7 because of an assumed initial vertical rise to 500 feet altitude. With high thrust the turn from this direction becomes costly. A continuous-burn constant thrust ascent to a 50 mile orbit altitude requires approximately 800 feet per second more acceleration than the two-burn transfer ellipses. This difference causes about a 7 percent difference in ascent stage weight and does not look attractive, particularly in view of the fact that the penalty becomes even more prohibitive if the ascent stage thrust-to-mass ratio is kept above 0.5 to cope with abort requirements. Accordingly, one of the transfer ellipse modes appears to be the best choice.

If the lunar orbit rendezvous technique is used, rendezvous will be required of either the ascent stage or the parent orbiting spacecraft. Generally rendezvous maneuvers have the same requirement as trajectory corrections and depend for their magnitude on the accuracy of the tracking and guidance systems. Assuming the guidance to be good, these maneuver requirements are small and installation of rendezvous systems in both the ascent spacecraft and the parent spacecraft is therefore not prohibitive in weight.

SURVEYOR

The Surveyor spacecraft (Figure 17) is being developed to land instruments on the moon to measure physical and chemical properties of the surface and to survey several prospective manned landing areas. This soft landing mission represents a significant advancement in spacecraft capability over the Ranger which is designed to photograph the lunar surface just prior to impacting at nearly the full velocity of its trans-lunar trajectory.

Mission Mode Selection

A descent directly to the lunar surface was selected as the Surveyor landing mode for reasons already presented in discussing unmanned lunar landing mission requirements. The direct descent, of course, minimizes propulsive requirements and permits continuous communication with the spacecraft. Since abort capability in the lunar vicinity is of little advantage for instrument package landings, the use of a lunar parking orbit was not considered.

The descent profile selected for the Surveyor spacecraft is illustrated in Figure 18. This landing is accomplished in two phases: the first phase reduces the spacecraft velocity from 8600 feet per second to about 400 feet per second at a distance of 28,000 feet above the lunar surface; the second phase reduces the velocity to 5 feet per second at a few feet above the surface.

Surveyor Propulsion Systems

Two on-board propulsion systems were selected for accomplishing the landing maneuvers and the mid-course correction maneuvers. A solid propellant motor provides the impulse for reducing the spacecraft velocity to 400 feet per second. A liquid propulsion system comprising three small engines will accomplish mid-course corrections, stabilize the vehicle during operation of the solid propellant motor, and accomplish the final reduction in velocity for the soft landing.

The solid propellant motor employs a spherical motor case and partially submerged nozzle (Figure 19), and weighs approximately 1,330 pounds. This motor, which represents a large portion of the total spacecraft weight at launch, is located on the center line of gravity of the vehicle and is carefully aligned so that the thrust vector will not produce significant turning moments. The motor delivers a thrust of approximately 8,000 pounds and operates for forty seconds during the lunar retro maneuver.

The liquid bi-propellant landing system employs nitrogen tetroxide as an oxidizer and blended hydrazine derivatives as the fuel. The propellant feed system comprises multiple positive-expulsion tanks, a high pressure gas expulsion system and three engines that can be throttled over a thrust range of 104 to 30 pounds. The engines are located on the landing legs of the spacecraft. Throttling is accomplished with a feed line control valve. The engine (shown in Figure 19) is equipped with a fixed injector and employs a combination of radiation and regenerative cooling.

An alternate engine configuration is also under development to increase the throttling range and reduce the feed pressure requirements, both of which will improve the payload capability of these spacecraft. The alternate engine employs a variable-area injector coupled to a variable-area cavitating-venturi control system. This combination permits a significant increase in the throttling range of the engine. Ablation and radiation cooling techniques are employed in this design.

APOLLO

The objective of the Apollo mission is manned lunar landing for exploration of the moon and subsequent safe return to Earth. The effort represents an annual expenditure averaging about one percent of our gross national product. This is less than the men of the United States spend on tobacco or the women spend on cosmetics.

The Apollo objective will be accomplished following extensive preparation of manufacturing shops, test and launch facilities, range complexes, and research laboratories. Each flight component will be extensively tested on the ground and in Earth-orbit flights to ascertain its capability of performing effectively during this mission. The exploration of the moon will culminate a nine-year scientific and engineering effort.

The mission, possibly the most extensive technological endeavor ever undertaken by man, occupies the principal efforts of perhaps 400,000 people, employed in government, university, and industrial installations located in many separated areas across the United States.

Mission Mode Selection

Three principal modes of performing the manned lunar mission were considered. These are illustrated in Figure 20. The modes proposed were direct flight, the use of Earth orbit rendezvous, and the use of a lunar orbit rendezvous.

In direct flight the entire lunar spacecraft assembly is lifted at one time. The earth orbit rendezvous involves assembling the spacecraft in an Earth orbit and requires rendezvous of separately launched parts of the spacecraft. It has the important advantage of permitting the lunar flight with a launch vehicle of roughly half the size of the direct flight launch vehicle.

The lunar orbit rendezvous mode involves the simultaneous transport of two spacecraft to a lunar orbit with subsequent separation of a lunar landing module. While the parent spacecraft remains in orbit this much smaller landing module descends to permit exploration of the lunar surface, then ascends and rejoins the parent spacecraft in lunar orbit. The parent spacecraft then returns to Earth.

Since this mode does not entail landing the entire spacecraft but only a small module specifically designed for that purpose it offers weight-saving advantages in the lunar spacecraft over either of the other modes. Of course, the use of lunar orbit rendezvous does not prohibit the use of Earth orbit rendezvous in the assembly of the spacecraft.

Scientists and engineers in NASA and other government agencies, as well as in the American industries and universities, devoted intensive study to the three main approaches discussed above. After extensive analysis it was concluded that for the manned landing mission the lunar orbital approach offered the greatest probability of success at a lower cost and on a faster schedule of accomplishment than either the direct ascent or the Earth orbital rendezvous techniques.

The direct flight method requires the largest launch vehicle although it requires no spacecraft rendezvous experience. The launch vehicle would need to be capable of lifting 150,000 pounds into trans-lunar trajectory (about 400,000 pounds in Earth orbit). Direct flight estimates indicated a later date for completion of the mission. Consequently, it was the first to be ruled out in this selection process.

The Earth orbit mode was studied in two versions. It was fairly evident that a method in which the spacecraft and a fueled escape vehicle would be put into orbit and then joined could not be accomplished with a logical division in the two payloads. The second alternative required rendezvous in Earth orbit between an unmanned tanker and a manned Apollo spacecraft including an unfueled injection stage. The tanker would fuel the injection stage while in orbit. After the fueling operation the manned spacecraft would be essentially the same as for the direct flight mode. Using this mode the mission could be accomplished by use of two similar launch vehicles with a capability of 200,000 pounds in orbit, thus avoiding delays incident to the development of the larger launch vehicle required for the lunar flight mode.

In the lunar orbit rendezvous mode the injected spacecraft weight could be reduced to approximately 90,000 pounds by eliminating the requirement for propulsion to land the entire spacecraft on the lunar surface. This injection weight corresponds to an Earth-orbit weight of 240,000 pounds, a launch vehicle capability similar to that required for the Earth-orbit rendezvous.

Comparison between the Earth-orbital rendezvous techniques brought out a number of pertinent points:

1) Lunar orbit rendezvous calls for one launch from Earth rather than two for Earth-orbital rendezvous.

2) Lunar orbital rendezvous requires about 6/10s of the payload weight in orbit as compared to the Earth orbit rendezvous. This smaller mass has a bearing on the lower costs.

3) Lunar orbit rendezvous permits optimization of the lunar landing spacecraft and also permits a corresponding optimization of the Earth re-entry module. In other words, by not combining both landing and re-entry capability in one craft, thus compromising one for the other, each can be tailored for its specific mission.

4) With all factors equal, rendezvous in Earth orbit would be somewhat less hazardous than in lunar orbit. However, there are off-setting factors in favor of the lunar orbit rendezvous: (a) In lunar orbit rendezvous the velocities of the rendezvous spacecraft are much lower than in the Earth orbit rendezvous (approximately 5,000 feet per second as compared with 26,000 feet per second); (b) In lunar orbit rendezvous both vehicles are manned and each can maneuver toward the other; (c) In lunar orbit rendezvous, the attachment of a 4,000 pound module is easier than executing the same maneuver with a module in excess of 200,000 pounds as in Earth orbit rendezvous.

5) Lunar orbit rendezvous offers the observation and reporting advantage of having two men descend while one stays in lunar orbit where he can observe and monitor the critical landing phase and report the operation back to Earth.

In addition to these advantages, the lunar orbit rendezvous technique was found to provide advantages in schedule and cost of development, while maintaining a high probability of mission safety and mission success.

Spacecraft Propulsion Systems

The Apollo spacecraft (Figure 21) comprises three major modules, which are identified as the Command Module (C/M), the Service Module (S/M), and the Lunar Excursion Module (L.E.M.).

The Command Module houses a three-man crew, subsystems to provide electrical power, communications, orientation, stabilization, and environmental control and life support for a 14-day mission. It serves as the control center of the spacecraft and is a blunt, conically shaped body, 13 feet in diameter at its base and 11 feet high (Figure 22). The skin is of brazed honeycomb steel construction. To protect the astronauts inside during re-entry into the Earth's atmosphere a heat shield made of a special material that ablates (boils to a gas) at extremely high temperatures is used.

The Service Module is primarily a propulsive module which provides first, the capability for making velocity corrections during the trans-lunar and trans-Earth journies; second, the energy for injection into lunar orbit and escape from lunar orbit after the landing mission is complete; and third, the necessary attitude control and maneuvering capability.

The Lunar Excursion Module is a two-stage vehicle designed with one stage for transporting two astronauts, along with scientific instruments and flight controls, from a lunar orbit to the lunar surface, and with the second stage for ascent and rendezvous with the Command Module-Service Module configuration.

Figures 23 and 24 illustrate the primary propulsive maneuvers that these modules are designed to accomplish. In addition to the normal maneuvers described, abort capability is also provided so that a safe return to Earth can be accomplished at any point during the mission should it be necessary to do so. To facilitate abort during launch, a Launch Escape Propulsion System is also provided (Figure 25). This solid propellant system is jettisoned during second stage burning in normal flight and since this lies outside the moon's sphere of activity this system will not be discussed in this paper.

The spacecraft propulsion systems selected for these applications are similar in many respects. All employ the Earth storable propellant combination of nitrogen tetroxide as the oxidizer and blended hydrazine derivatives as the fuel. The use of these propellants for the Apollo spacecraft systems was accepted because of extensive exploratory research in development programs which had preceded this requirement. The Earth-storable propellants alleviate some of the problems of storage during the 10-day duration mission to the moon and return. Figure 26 illustrates the temperature-differential problems introduced by use of the higher-performance cryogenic propellants. This selection was a case of sacrificing performance for assurance that the propulsion system could be developed successfully within the schedule restraints. Simple pressurized propellant systems serve to pump the fuel and oxidizer to the engine combustion chambers. Simplicity and reliability are emphasized in the system designs. Characteristic of these systems are redundant valves, regulators, and other feed system components where substantial gains in reliability can be realized by use of redundancy without significant system complication or weight penalties. In the primary propulsion systems, single engines are employed. Redundant engines are provided in the control systems.

North American Aviation Space and Information Systems Division is the principal contractor to NASA's Manned Spacecraft Center in Houston, Texas, for developing and building the Command and Service Modules. The Grumman Aircraft Engineering Corporation at Long Island, New York, is building the Lunar Excursion Module. Each contractor has subcontracted the development and fabrication of the thrusters, tanks, and other propulsion system components.

Service Module

The primary function of the Service Propulsion Subsystem is to provide thrust for mid-course corrections, for entry into the lunar orbit, for trans-Earth injection from lunar orbit, and for emergency maneuvers. On Earth-orbital missions, it also provides retro-thrust for entry into the Earth's atmosphere.

The Service Module contains two propulsion systems; the Service Propulsion Sybsystem for the principal maneuvers of lunar orbit entry and departure, and the Service Module Reaction Control Subsystem (Figure 27). This module also houses fuel cells for electrical power generation and contains antennas and other associated equipment that is not required in the Command Module for re-entry purposes.

The Service Propulsion Subsystem comprises four cylindrical propellant tanks, two each for fuel and oxidizer, a helium pressurization subsystem and an engine that delivers nearly 22,000 pounds of thrust (Figure 28).

The engine, shown in Figure 29, is being developed by Aerojet-General Corporation at Sacramento, California. This engine has multiple restart capability. It operates at a pressure of about 100 pounds per square inch, absolute, and is cooled by a combination of ablation and radiation techniques. The combustion chamber and nozzle for a short distance down-stream of the throat is constructed of ablative glass-fiber-reinforced resin. The remainder of the exhaust nozzle, which extends to an area ratio of 60, is of refractory metal. The portion just behind the throat, at a point exposed to temperatures of 2200°F, is made of a columbium alloy. The aft end of the nozzle is made of titanium. This nozzle, exposed to space, is cooled by radiation to the space environment.

The engine is mounted in the vehicle so that it can be gimballed to accomplish thrust vector control.

The redundant propellant valve assembly and gimballed thrust mount are shown in Figure 30.

The Service Module Reaction Control Subsystem will provide pitch, roll, yaw and maneuver control for the spacecraft during the entire flight from injection into the lunar trajectory until Earth re-entry. In Earth-orbit missions it is also used as a redundant retro-thrust system for returning to Earth.

The Service Module Reaction Control Subsystem, located at the forward end of the Service Module, comprises four small propulsion modules. Only one opposite pair of these is required to accomplish the mission. These modules, located at 90 degrees intervals around the periphery of the stage, each contain a complete pressurized propulsion system with four engines, all again exhausting at about 90 degrees to each other. This subsystem is designed for a wide range of duty cycles from pulse type operation to continuous operation for several hundred seconds. These 100 pound thrust control engines, under development by the Marquardt Corporation of Van Nuys, California are constructed from high temperature metals and are cooled entirely by radiation (Figure 31). Propellant flow is controlled by two electrically actuated valves located just up-stream of the engine injector. The propellant is contained in two cylindrical propellant tanks that employ an elastomer bladder for positive expulsion control. This means of positively displacing the propellants in the tanks permits the subsystem to operate in the "zero-g" environment. Helium is employed in the pressurization system.

Lunar Excursion Module

The Lunar Excursion Module, which will carry the two astronauts with their life support equipment and scientific equipment to the lunar surface and return them to lunar orbit for rendezvous with the spacecraft, contains three propulsion subsystems; two in the ascent stage, and one in the descent stage. (Figure 32)

The descent stage propulsion subsystem differs from that previously described for the Service Propulsion Subsystem in that it is smaller and the engine throttles over a broad range of thrust. The Lunar Excursion Module descent engine has thrust variable from 10,500 pounds to 1,050 pounds and is gimballed to provide thrust vector control.

The basic engine construction is similar to that of the Service Propulsion System. An ablative combustion chamber-nozzle section is employed with a radiation cooled nozzle extension. The propellant flow rate is controlled to accomplish the throttling.

Two engines, employing different basic control concepts, are being developed simultaneously to assure that the throttling requirements can be met satisfactorily. One concept, under development by the Space Technology Laboratories of Los Angeles, California, employs variable-area cavitating-venturii for controlling the propellant flow rate to the engine and variable-area injection ports that are slaved to the variable-area cavitating-venturi flow control system (Figure 33). This system is like the system being developed alternatively for the Surveyor spacecraft. An alternate approach, under development by the Rocketdyne Division of North American Aviation, Incorporated, Los Angeles, California, employs a constant-area injector with a conventional feed line throttling valve (Figure 34). Helium is injected and mixed with the propellants in the low-thrust regime to provide injection velocities suitable for obtaining good combustion efficiency and stable system operations. One of these two throttling methods will soon be selected for application to the Lunar Excursion Module descent stage.

The ascent stage employs a 3,500 pound constant-thrust engine for returning the Lunar Excursion Module to the orbiting spacecraft. This engine, under development by the Bell Aerosystems Company of Buffalo, New York, operates at a pressure of about 110 pounds per square inch absolute and employs ablative cooling throughout. Figure 35 illustrates the engine configuration selected. Both the descent and ascent propulsion systems employ multiple propellant tanks, redundant valves and regulators and a helium pressurization system.

The Lunar Excursion Module reaction control system is active during both lunar descent and ascent. It is located in the ascent stage. It may be used to supplement the control provided by gimballing the descent engine during landing. It is the only control available during the ascent and rendezvous phase since the primary engine is fixed and has no means of controlling its thrust vector. The fixed engine design was influenced by the compact stage design which places the main engine very close to the center of gravity of the ascent stage where thrust vector by gimballing or other means would not be effective.

The reaction control system is very similar to that under development for the Service Module Reaction Control System, infact, the same 100 pound thrust radiation cooled engine will be used in both applications. Installation differences occur in the mounting of the engines and in the propellant feed system configuration. Two complete propulsion systems comprising eight engines each, make up the control system. Only one system is required for mission success. The propellant tanks also employ an elastomer bladder for positive expulsion and utilize a helium system for propellant expulsion.

The Apollo Mission

The Apollo mission to land men on the moon will begin at the John F. Kennedy Space Center in Florida. Three astronauts will be launched, seated in the Command Module, approximately 320 feet above the base of the Saturn V launch vehicle (Figure 36). The five F-1 engines of the Saturn V first stage, which generate 7½ million pounds of thrust, will burn for 2½ minutes. After separation of the first stage, the second stage will ignite (Figure 37).

Its 5 engines generate a total of 1,000,000 pounds of thrust. After 6½ minutes burning time, the third stage will be ignited (Figure 38). Its single engine provides 200,000 lbs. of thrust. This engine will burn initially about 2-3/4 minutes to hurl the spacecraft into earth orbit, and then will be shut down (Figure 39). In this parking orbit the spacecraft will be checked out by the astronauts and by ground control through telemetry. After several orbits and after its trajectory has been accurately ascertained by ground control, the third stage engine will again ignite for approximately five minutes to accelerate the spacecraft on a trajectory toward the moon (Figure 40).

At this point, the adaptor surrounding the Lunar Excursion Module will be separated (Figure 41). The Command Module and Service Module will also be separated leaving the Lunar Excursion Module attached to the third stage. The astronauts will then accomplish a turn around maneuver (Figure 42) and dock the Command Module nose-to-nose with the Lunar Excursion Module (Figure 43), after which the third stage will be separated. After this short-coupled trans-lunar rendezvous, the configuration will remain intact during the remainder of the flight to the lunar orbit. In-flight corrections and deceleration to place the spacecraft in lunar orbit will be accomplished by the 22,000 lb. thrust Service Module engine (Figure 44). The lunar orbit will be between 80 and 100 miles above the moon surface.

Two of the astronauts will then climb through the hatch of the Command Module into the Lunar Excursion Module (Figure 45). The descent engine will ignite and burn for ½ minute (Figure 46) to place the Lunar Excursion Module in an elliptical orbit which dips down to within 10 miles of the lunar surface where the astronauts can observe the surface for a suitable landing site (Figure 47). Speed of the Lunar Excursion Module with respect to the lunar surface will be about 4,000 miles per hour. Deceleration and landing of the Lunar Excursion Module will be accomplished by use of the Lunar Excursion Module throtteable engine which will have a thrust range from 1,000 to 10,000 lbs. (Figure 48).

While on the moon, two astronauts will alternate in leaving the Lunar Excursion Module to explore the surface and make scientific measurements (Figure 49). The total length of stay will be about 24 hours.

Launch from the moon will be accomplished by separating the landing stage of the Lunar Excursion Module (Figure 50). Using the ascent stage 3500 lb. thrust engine the Lunar Excursion Module will be accelerated to about 4,000 miles per hour at an altitude of 10 miles. Radars aboard both the Command Module and the Lunar Excursion Module will track each other and the Lunar Excursion Module will be used to make course corrections to insure rendezvous (Figure 51). Both modules will have the capability of making rendezvous and docking.

After the astronauts have returned to the Command Module through the hatch, the Lunar Excursion Module will be detached and left as a satellite in lunar orbit. The 22,000 lb. thrust Service Module engine will be used to accelerate the Command Module toward the earth (Figure 52); final corrections to hit the reentry corridor will be made with the Service Module engine after which the Service Module will be separated (Figure 53). Through use of auxiliary attitude control rocket engines, the Command

Module will position itself for reentry into the earth atmosphere (Figure 54) and at a suitable altitude deploy parachutes for landing on the earth surface (Figure 55).

The time for the entire mission will be between 8 and 10 days.

CONCLUDING REMARKS

It will be a historic day when these men come back from the moon but that will be a prelude, not a finale. We should again contemplate the sage prophesies of Tsiolkovsky, who said in the last century, "Man will not always remain on earth; the pursuit of light and space will lead him to penetrate the bounds of the atmosphere, timidly at first, but in the end to conquer the whole of solar space".

The successful accomplishment of these lunar missions will open the way for wider exploration of space and our solar system, leading to a more complete knowledge of our universe and a more profound understanding of the universal laws. The importance of such revelations cannot now be fully foreseen.

I can assure you on behalf of all in the United States who are engaged in the search for new knowledge in the new domain of space, that it is our paramount hope and purpose that the exploration of space will contribute to the achievement of genuine international cooperation and an uplifting of human culture through science and technology. Two of the stated objectives of the Space Act, which created the National Aeronautics and Space Administration, are: 1) to expand human knowledge and understanding through scientific and experimental study of the solar system and the space environment, and 2) to assure prompt dissemination of the new knowledge of space phenomena and technology throughout the scientific, technical and business communities of the world for the maximum benefit of all mankind. I trust that this dissertation on a very limited subject has contributed something to these purposes.

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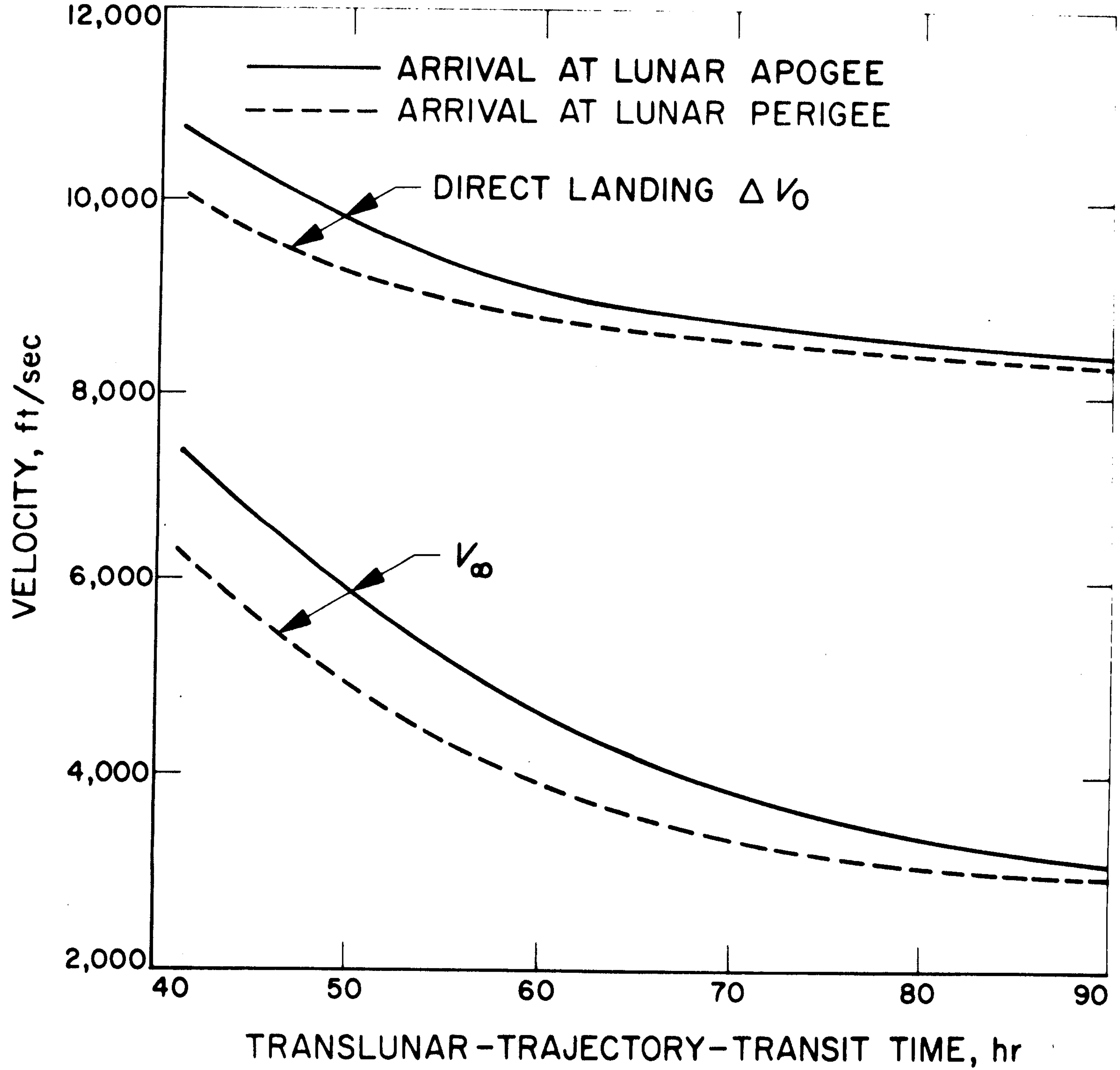
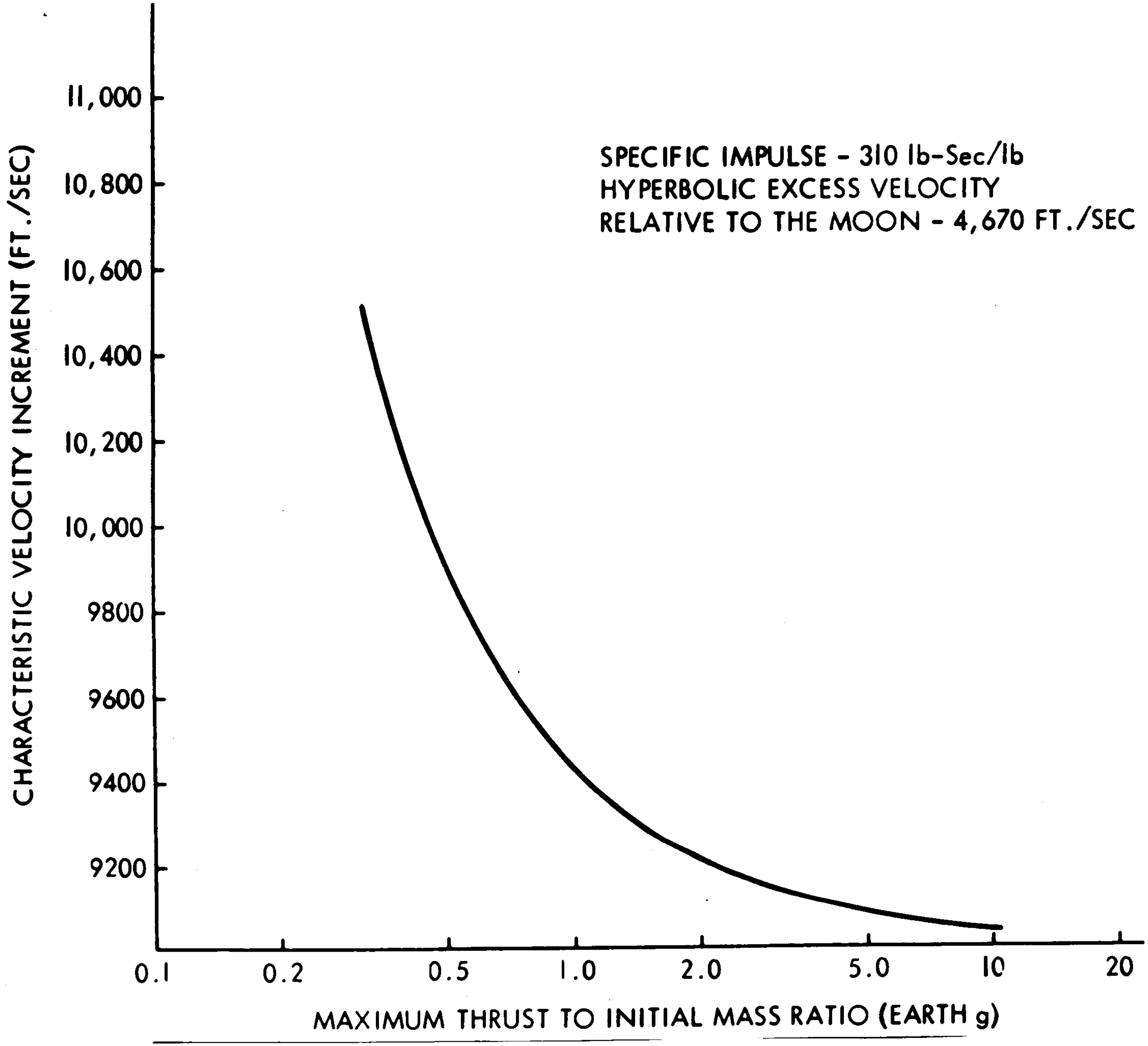
9. Thomas, David F., and Bird, John D.: A Parametric Investigation of the Lunar-Orbit-Rendezvous Scheme, NASA TN D-1623. April 1963.

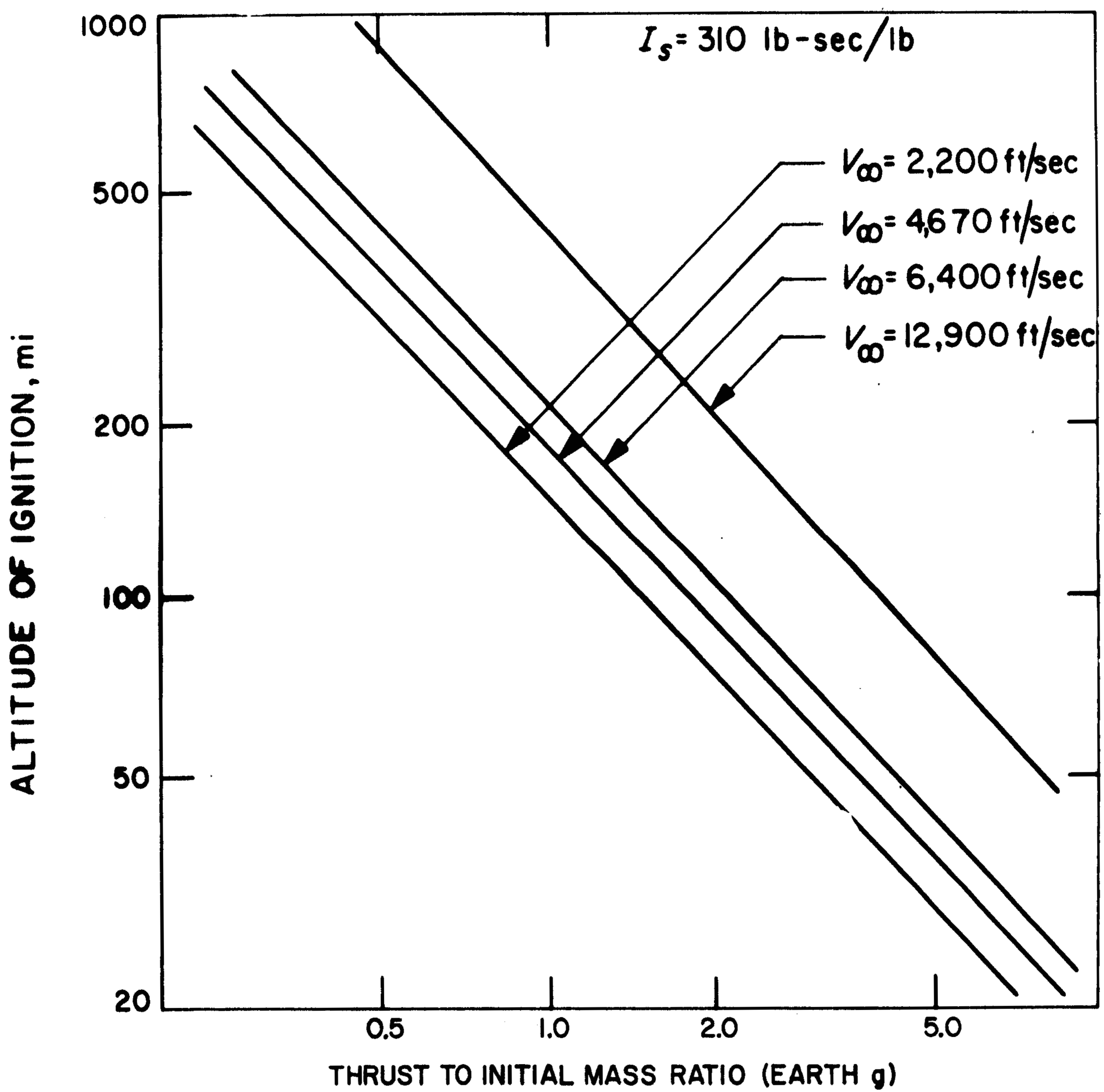
LIST OF ILLUSTRATIONS



ARRIVAL AT LUNAR APOGEE
ARRIVAL AT LUNAR PERIGEE
DIRECT LANDING ΔV_0
V_∞
VELOCITY, ft/sec
12,000
10,000
8,000
6,000
4,000
2,000
40
50
60
70
80
90
TRANSLUNAR-TRAJECTORY-TRANSIT TIME, hr

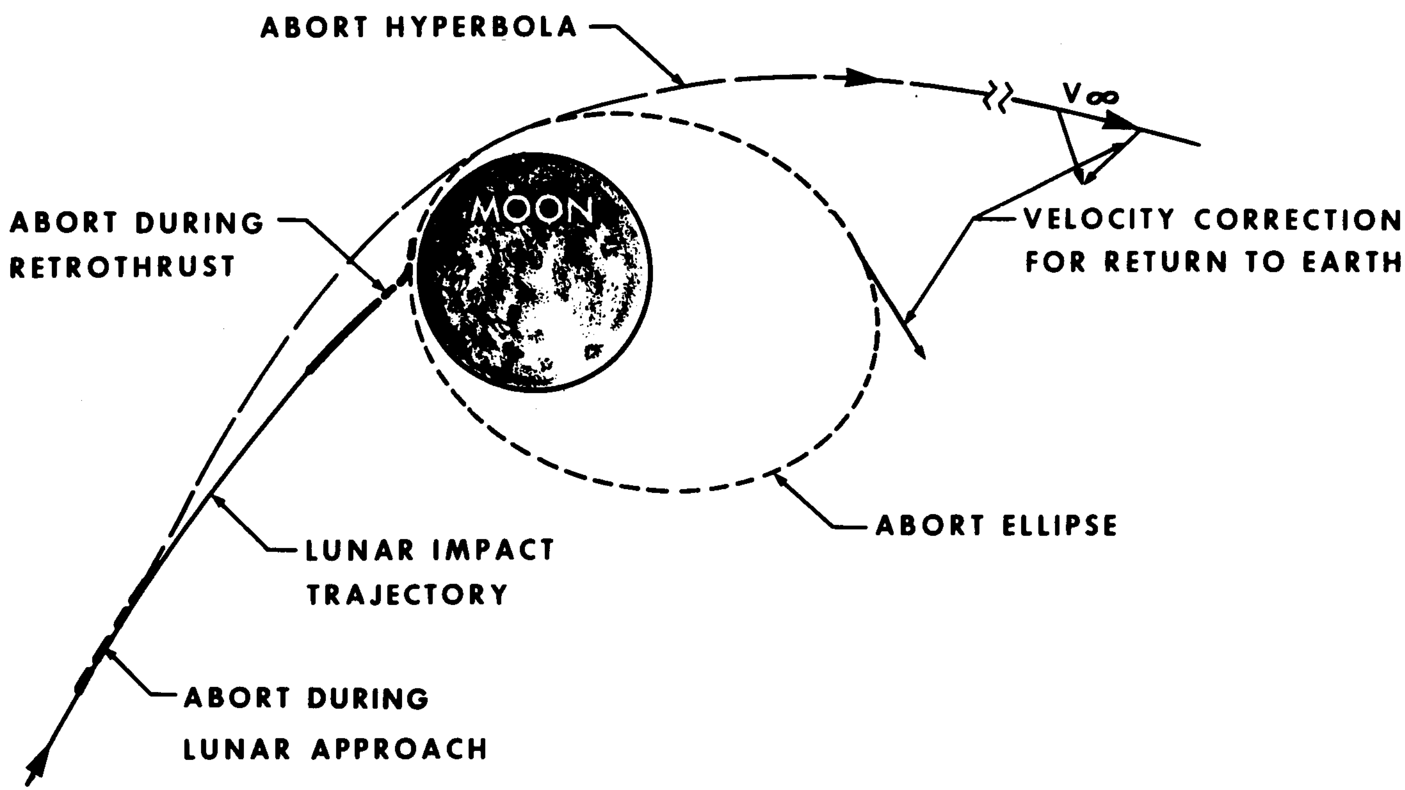


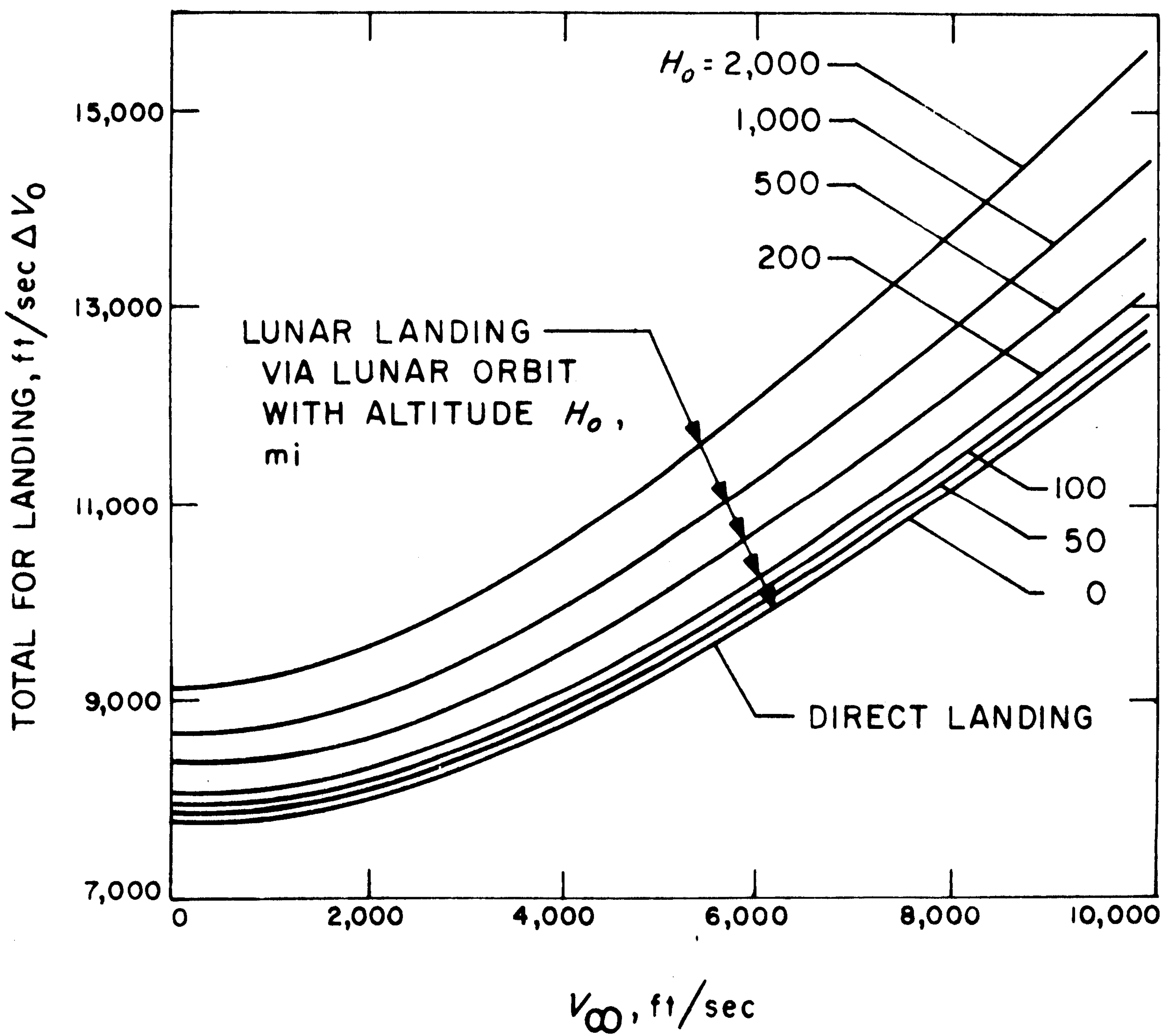
CHARACTERISTIC VELOCITY INCREMENT (FT./SEC)
11,000
10,800
10,600
10,400
10,200
10,000
9800
9600
9400
9200
SPECIFIC IMPULSE - 310 lb-Sec/lb
HYPERBOLIC EXCESS VELOCITY
RELATIVE TO THE MOON - 4,670 FT./SEC
0.1
0.2
0.5
1.0
2.0
5.0
10
20
MAXIMUM THRUST TO INITIAL MASS RATIO (EARTH g)



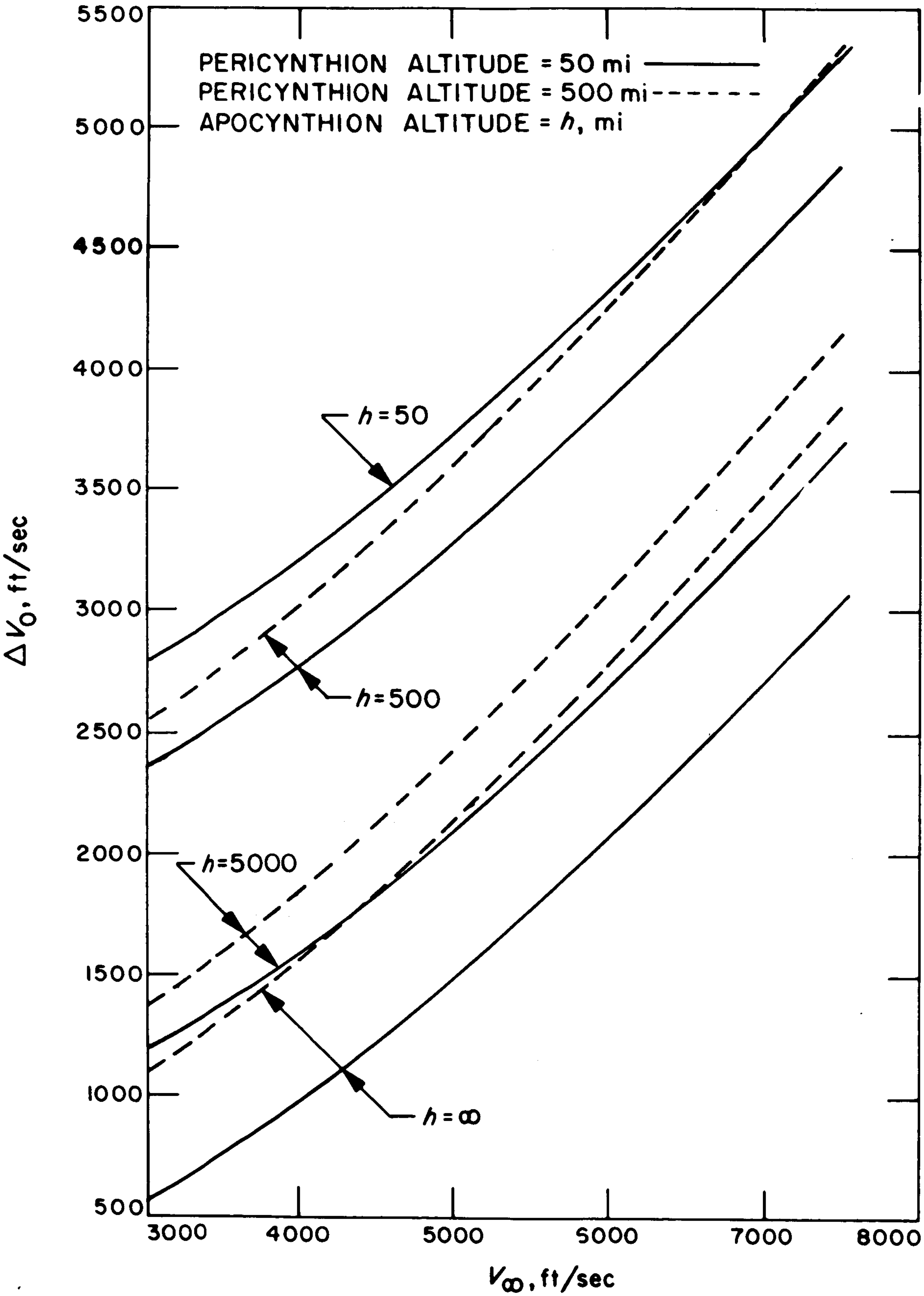
ALTITUDE OF IGNITION, mi
1000
500
200
100
50
20
I_s = 310 lb-sec/lb
V_∞ = 2,200 ft/sec
V_∞ = 4,670 ft/sec
V_∞ = 6,400 ft/sec
V_∞ = 12,900 ft/sec
0.5
1.0
2.0
5.0
THRUST TO INITIAL MASS RATIO (EARTH g)

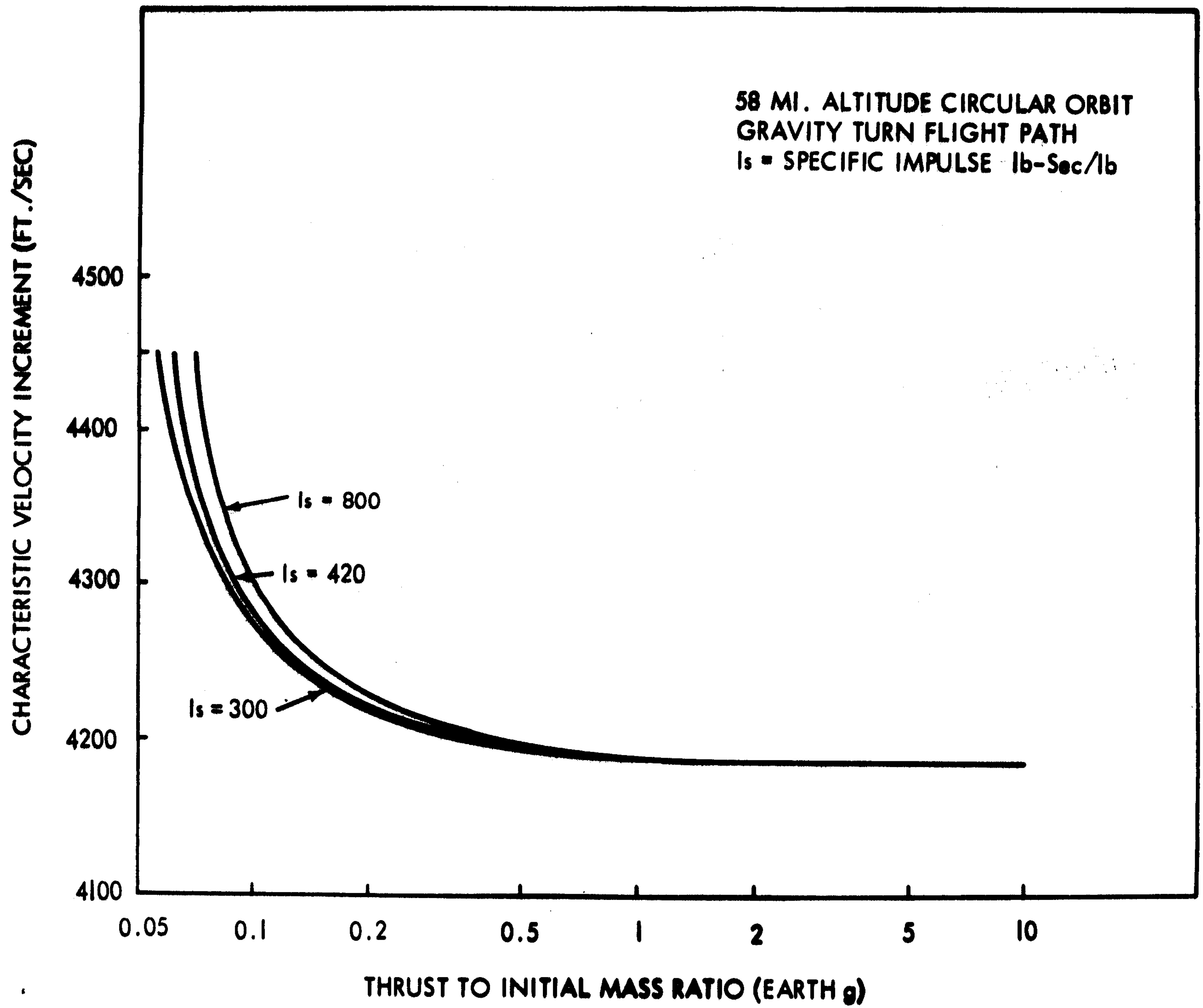
TYPICAL TRAJECTORY CHARACTERISTICS FOR ABORT DURING A DIRECT LANDING.



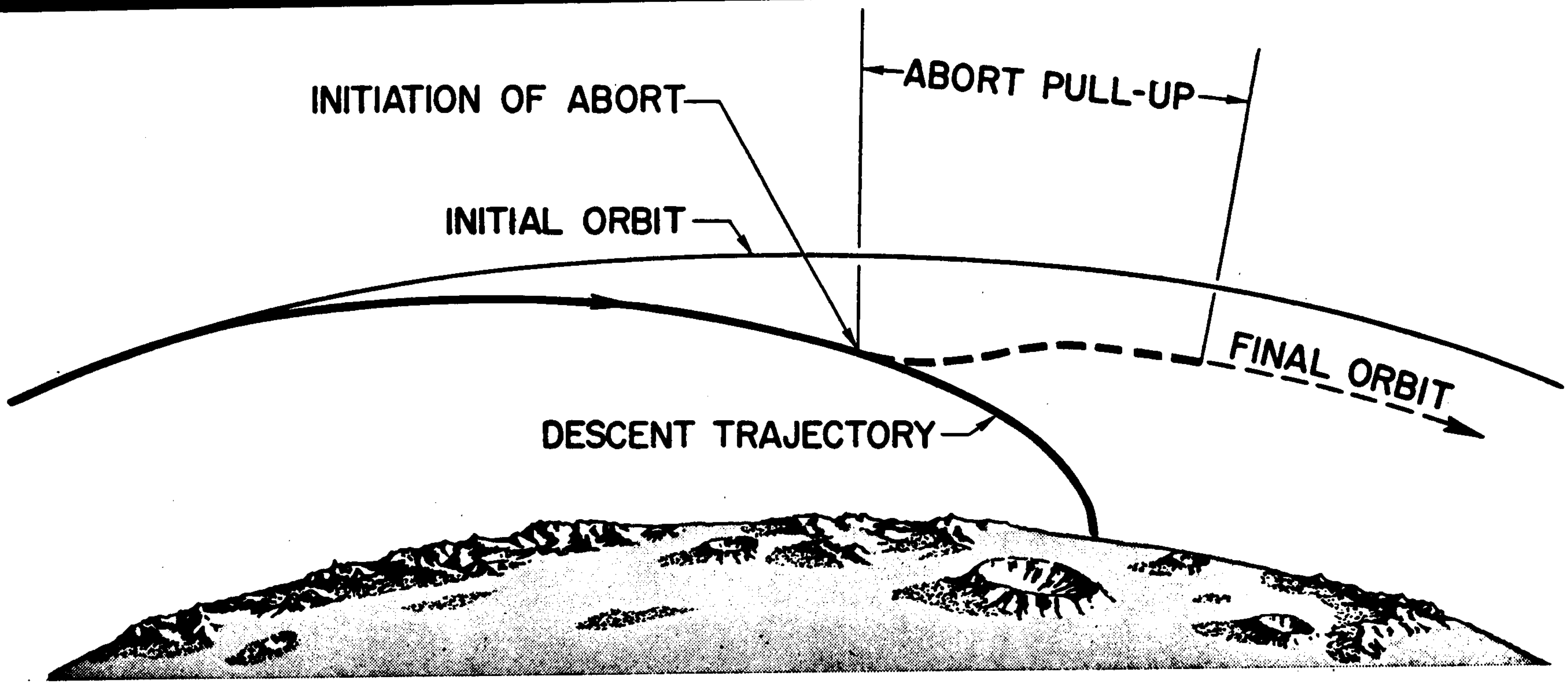


H_o = 2,000
1,000
500
200
100
50
0
LUNAR LANDING
VIA LUNAR ORBIT
WITH ALTITUDE H_o,
mi
DIRECT LANDING
TOTAL FOR LANDING, ft/sec ΔV_0
15,000
13,000
11,000
9,000
7,000
0
2,000
4,000
6,000
8,000
10,000
V_∞, ft/sec

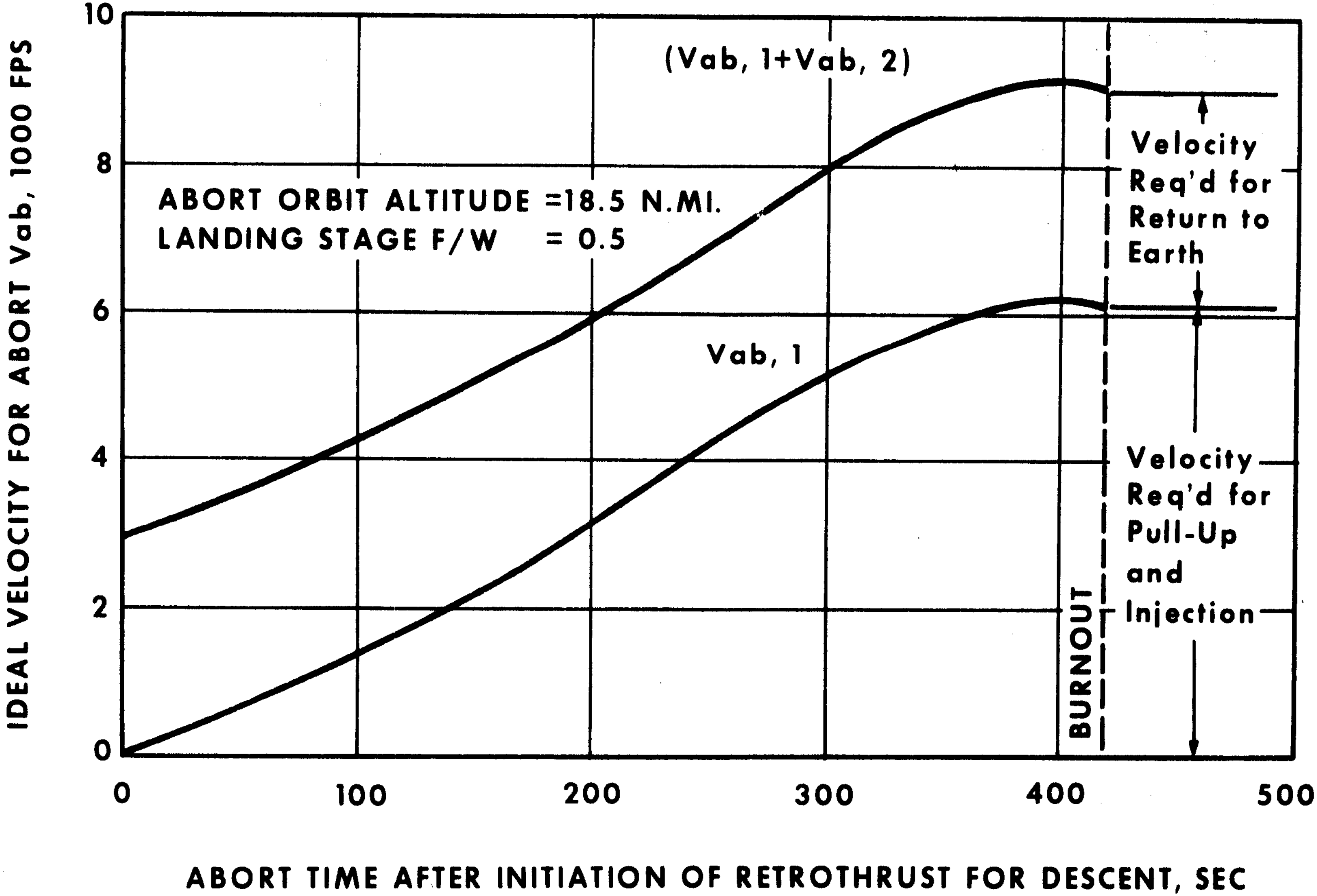

PERICYNTHION ALTITUDE = 50 mi
PERICYNTHION ALTITUDE = 500 mi
APOCYNTHION ALTITUDE = h, mi
h = 50
h = 500
h = 5000
h = ∞
5500
5000
4500
4000
3500
3000
2500
2000
1500
1000
500
3000
4000
5000
5000
7000
8000
ΔV0, ft/sec
V∞, ft/sec

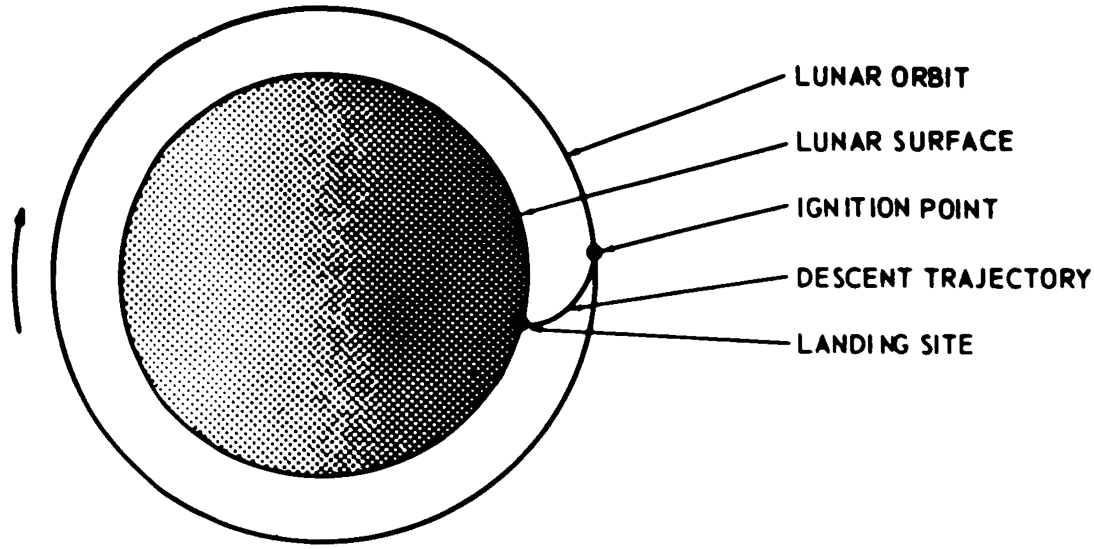

58 MI. ALTITUDE CIRCULAR ORBIT
GRAVITY TURN FLIGHT PATH
Is = SPECIFIC IMPULSE lb-Sec/lb
CHARACTERISTIC VELOCITY INCREMENT (FT./SEC)
4500
4400
4300
4200
4100
Is = 800
Is = 420
Is = 300
0.05
0.1
0.2
0.5
1
2
5
10
THRUST TO INITIAL MASS RATIO (EARTH g)

TYPICAL FLIGHT PROFILE FOR ABORT DURING A PARKING ORBIT LANDING

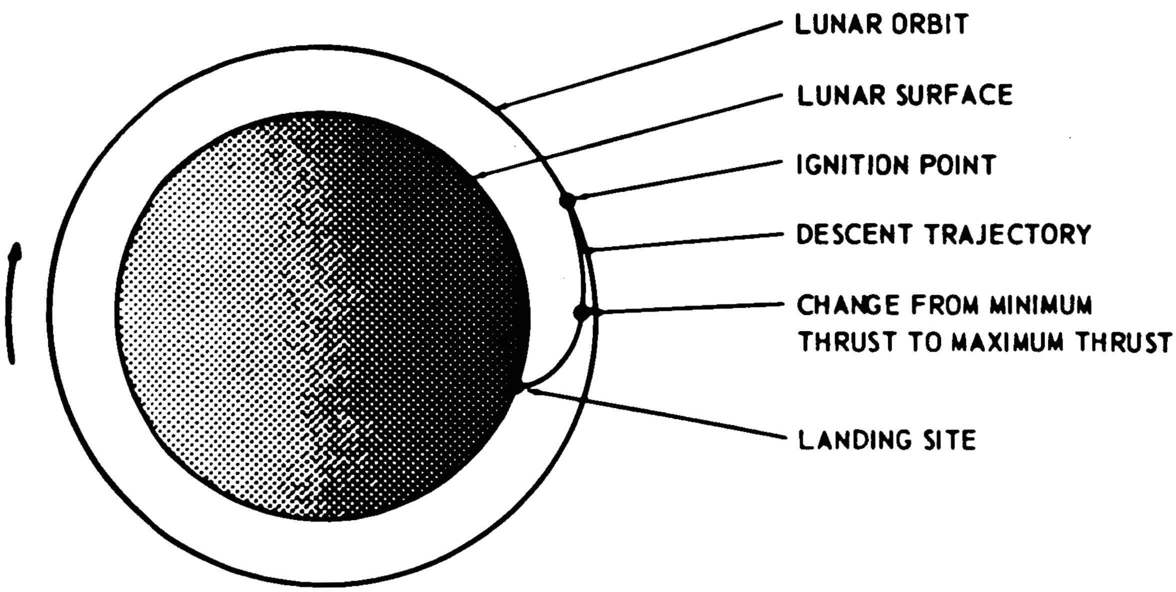


VELOCITY REQUIREMENTS FOR ABORT DURING A PARKING ORBIT LANDING

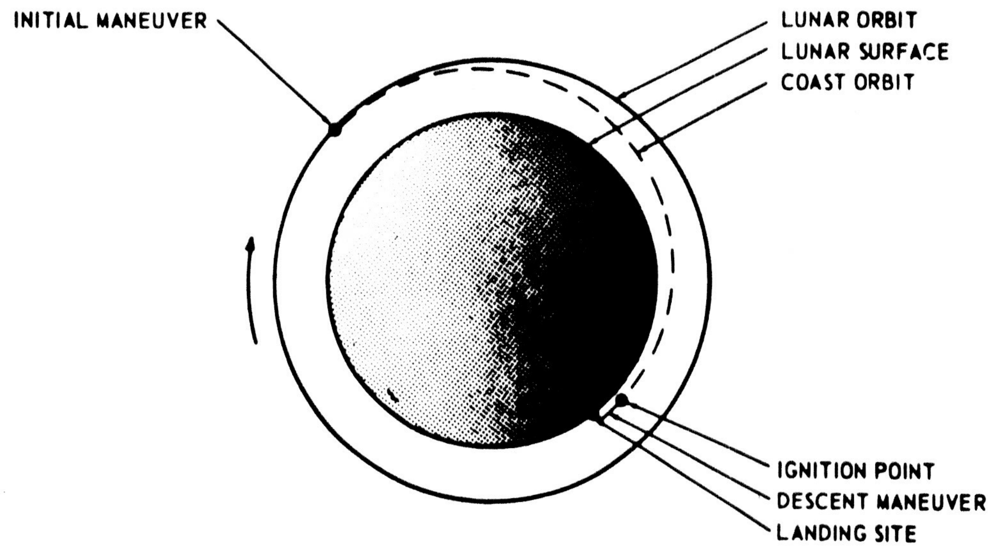




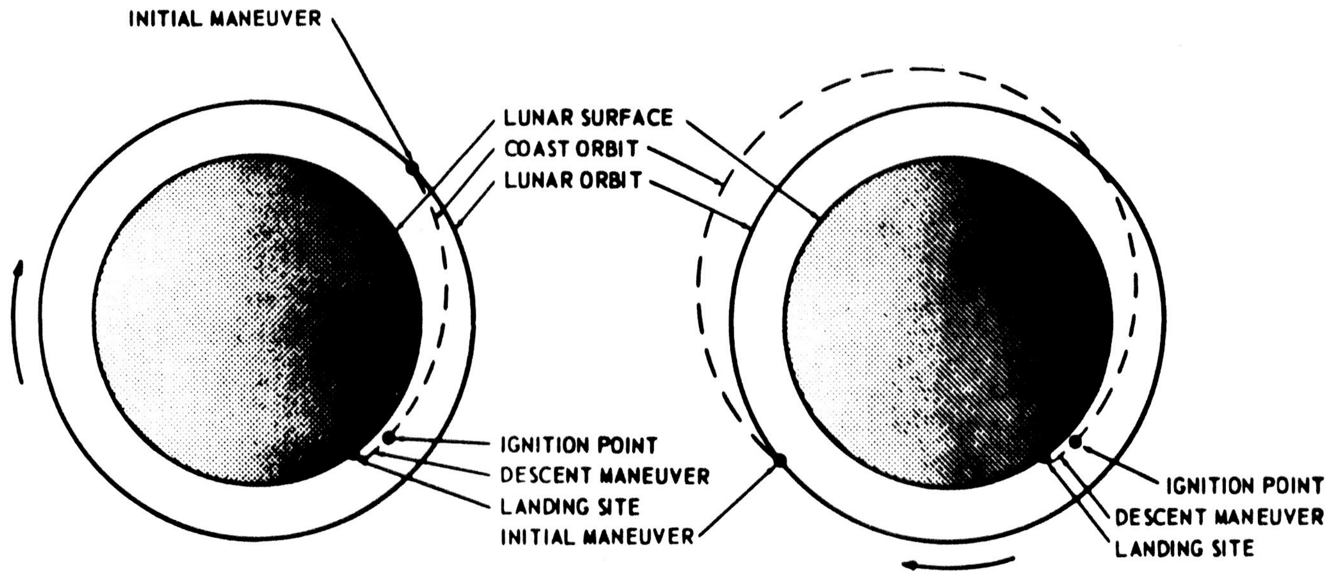
a. CONTINUOUS CONSTANT-THRUST DESCENT



b. CONTINUOUS VARIABLE-THRUST DESCENT



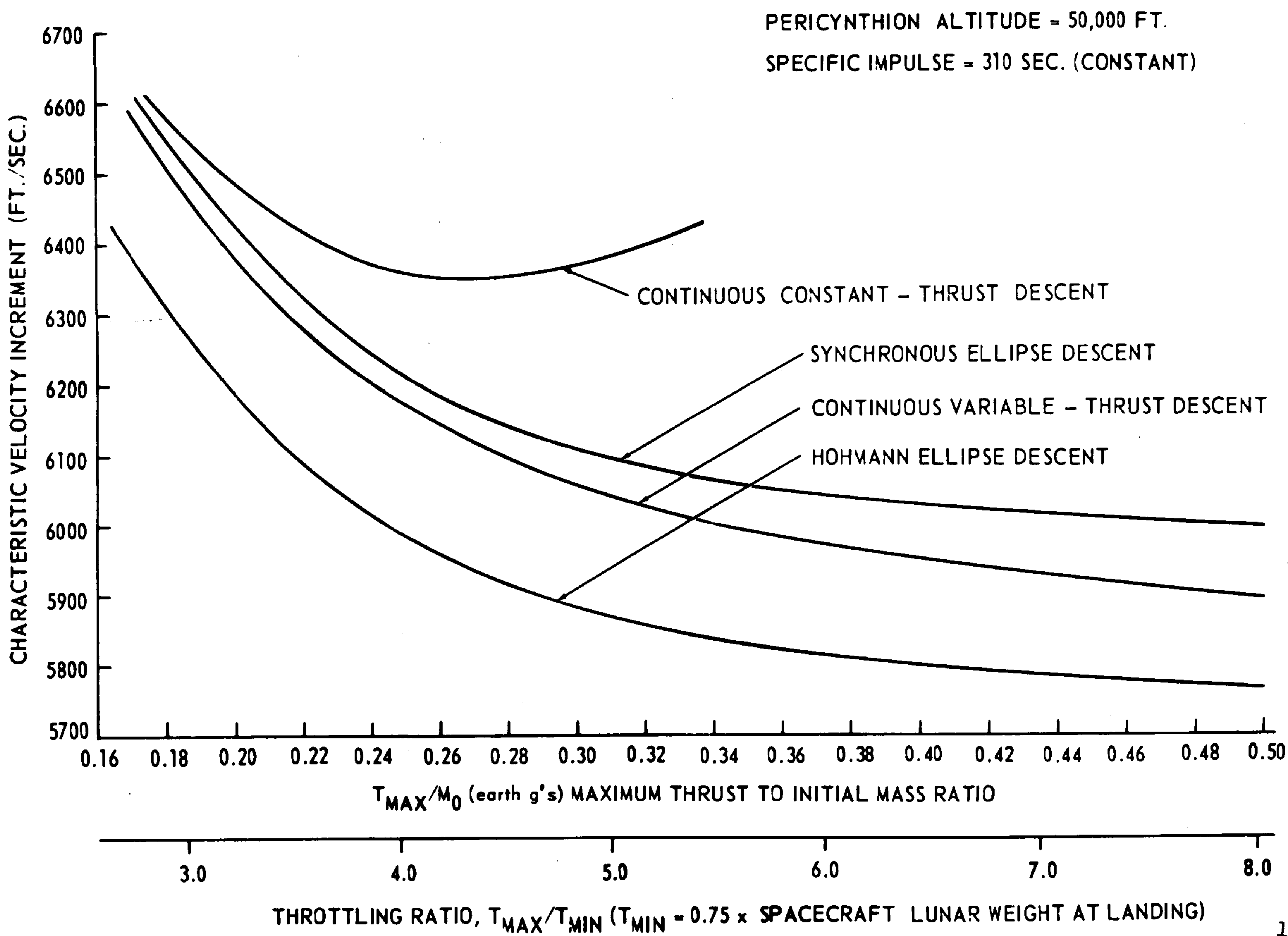
c. HOHMANN ELLIPSE DESCENT



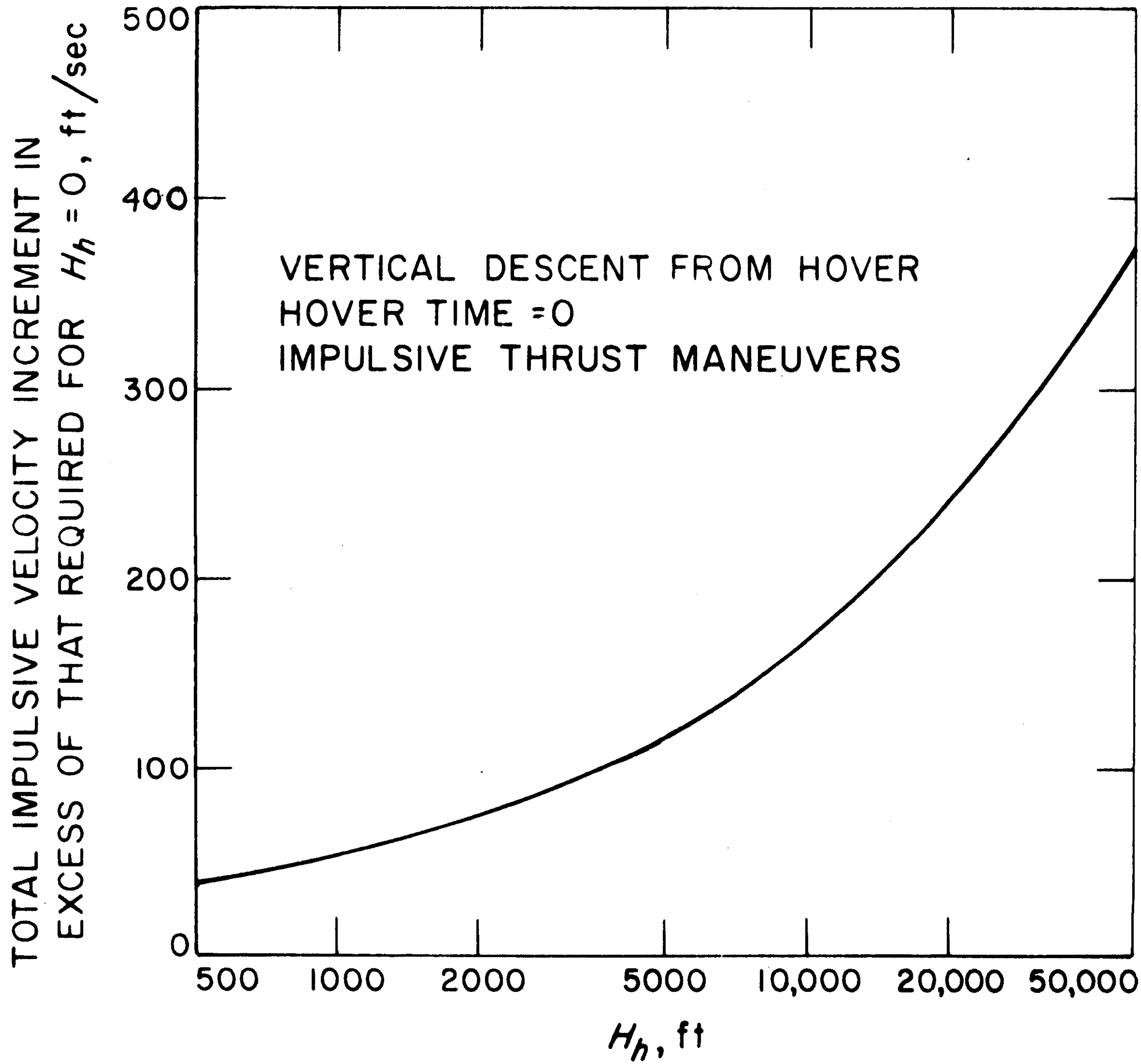
90° COAST

270° COAST.

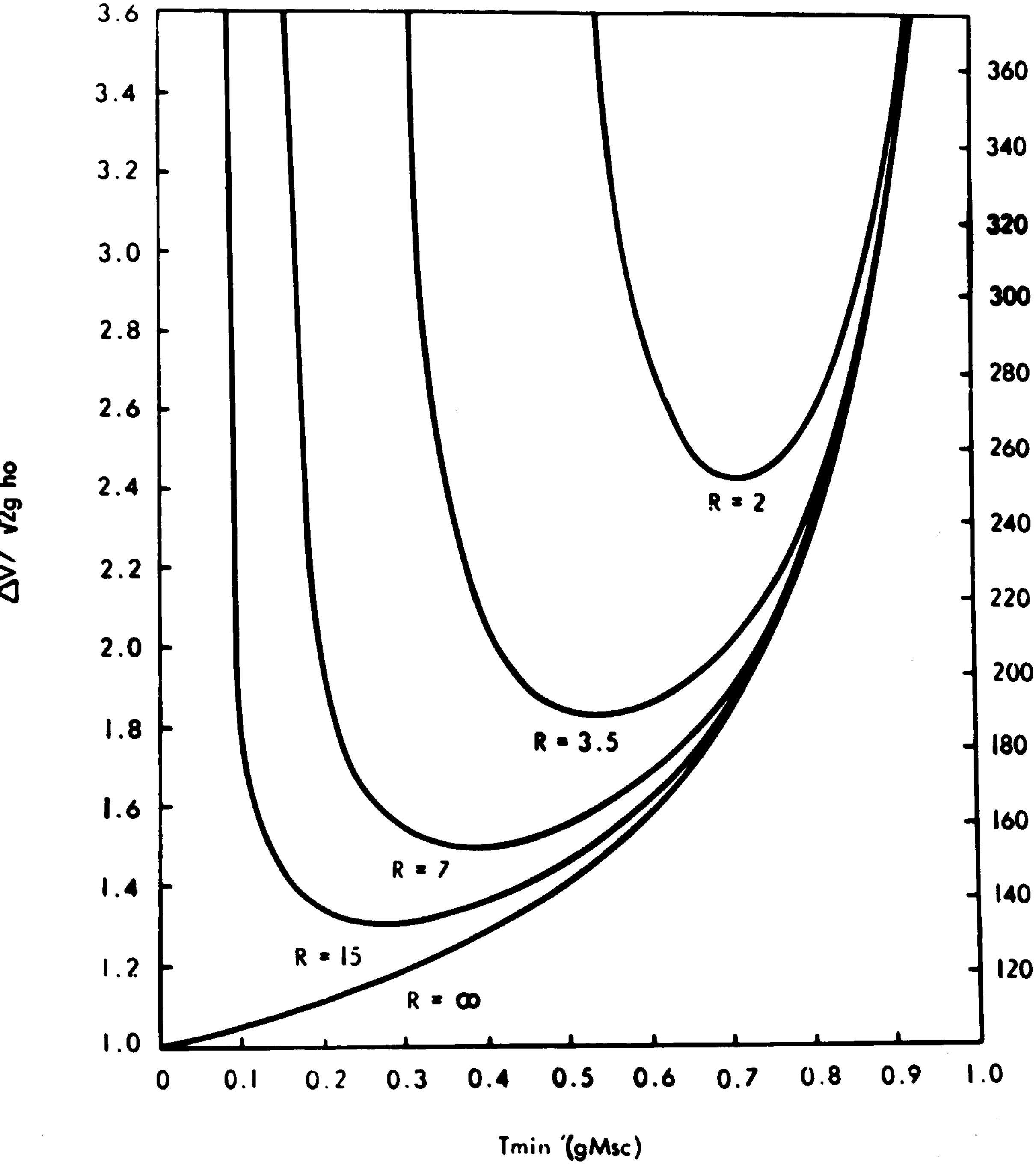
d. SYNCHRONOUS ELLIPSE DESCENT



PERICYNTHION ALTITUDE = 50,000 FT.
SPECIFIC IMPULSE = 310 SEC. (CONSTANT)
CHARACTERISTIC VELOCITY INCREMENT (FT./SEC.)
6700
6600
6500
6400
6300
6200
6100
6000
5900
5800
5700
CONTINUOUS CONSTANT – THRUST DESCENT
SYNCHRONOUS ELLIPSE DESCENT
CONTINUOUS VARIABLE – THRUST DESCENT
HOHMANN ELLIPSE DESCENT
0.16 0.18 0.20 0.22 0.24 0.26 0.28 0.30 0.32 0.34 0.36 0.38 0.40 0.42 0.44 0.46 0.48 0.50
T_{MAX}/M_0 (earth g's) MAXIMUM THRUST TO INITIAL MASS RATIO
3.0 4.0 5.0 6.0 7.0 8.0
THROTTLING RATIO, T_{MAX}/T_{MIN} (T_{MIN} = 0.75 x SPACECRAFT LUNAR WEIGHT AT LANDING)

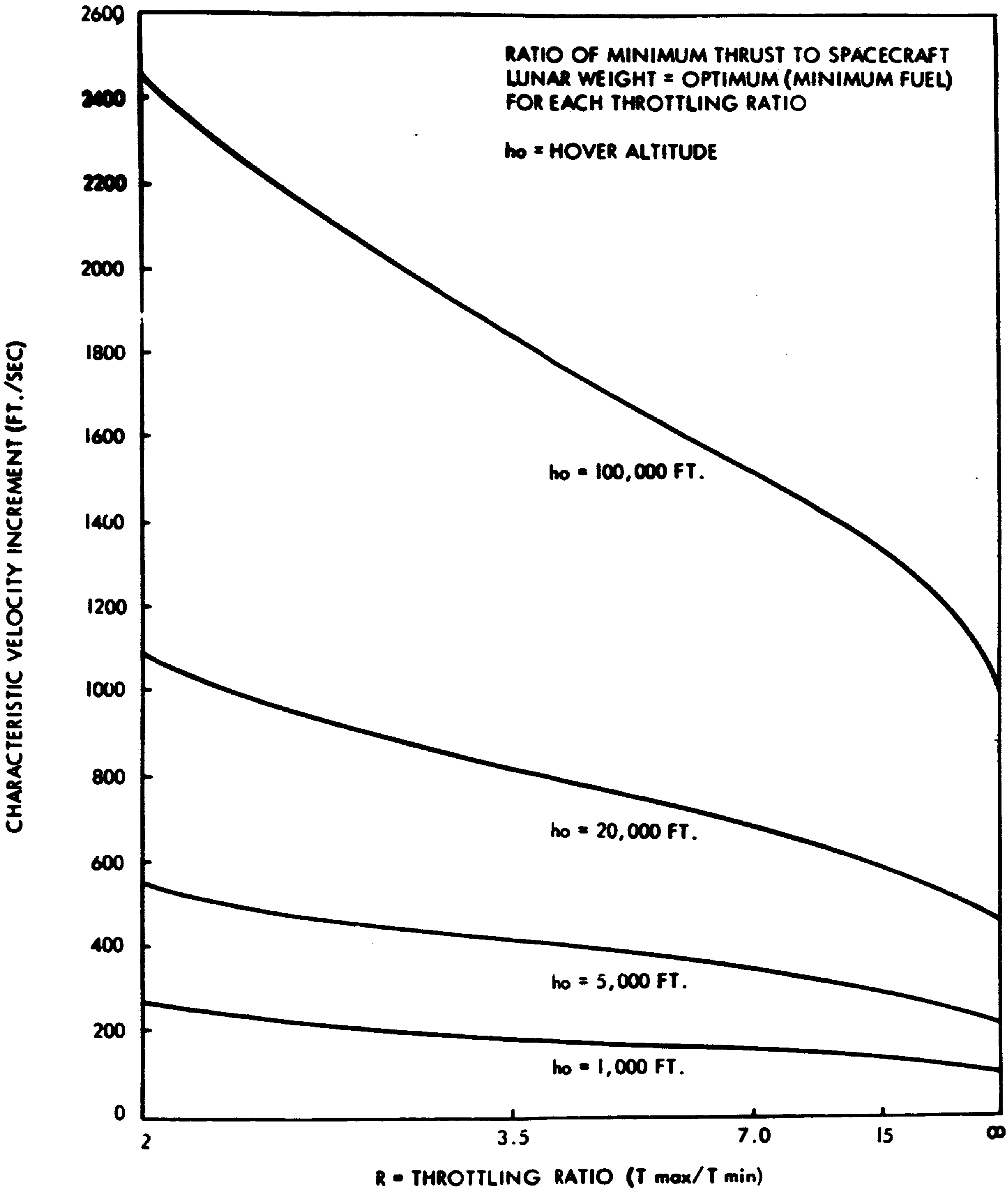


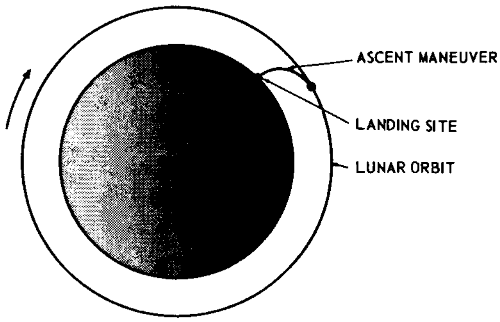
VERTICAL DESCENT FROM HOVER
HOVER TIME = 0
IMPULSIVE THRUST MANEUVERS
TOTAL IMPULSIVE VELOCITY INCREMENT IN EXCESS OF THAT REQUIRED FOR $H_h = 0$, ft/sec
0
100
200
300
400
500
500
1000
2000
5000
10,000
20,000
50,000
H_h, ft



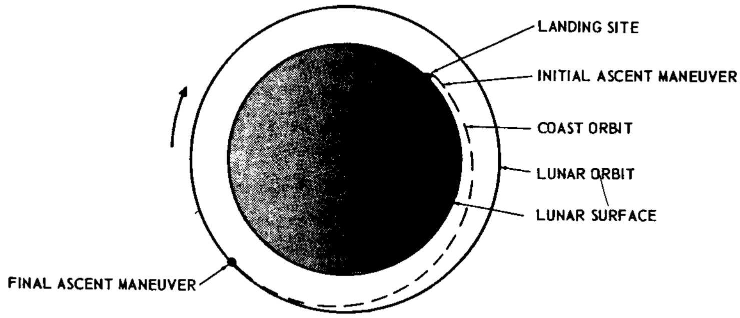
ΔV = CHARACTERISTIC VELOCITY INCREMENT
ho = HOVER ALTITUDE
g = LUNAR ACCLERATION OF GRAVITY
T min = MINIMUM THRUST
Msc = AVERAGE SPACECRAFT MASS DURING DESCENT
R = THROTTLING RANGE T max/T min

FIGURE 13

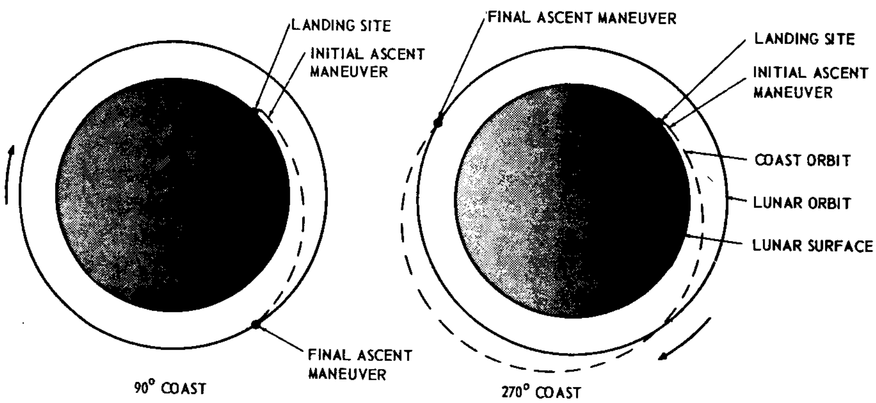

RATIO OF MINIMUM THRUST TO SPACECRAFT LUNAR WEIGHT = OPTIMUM (MINIMUM FUEL) FOR EACH THROTTLING RATIO
ho = HOVER ALTITUDE
ho = 100,000 FT.
ho = 20,000 FT.
ho = 5,000 FT.
ho = 1,000 FT.
CHARACTERISTIC VELOCITY INCREMENT (FT./SEC)
2600
2400
2200
2000
1800
1600
1400
1200
1000
800
600
400
200
0
2
3.5
7.0
15
∞
R = THROTTLING RATIO (T max/T min)



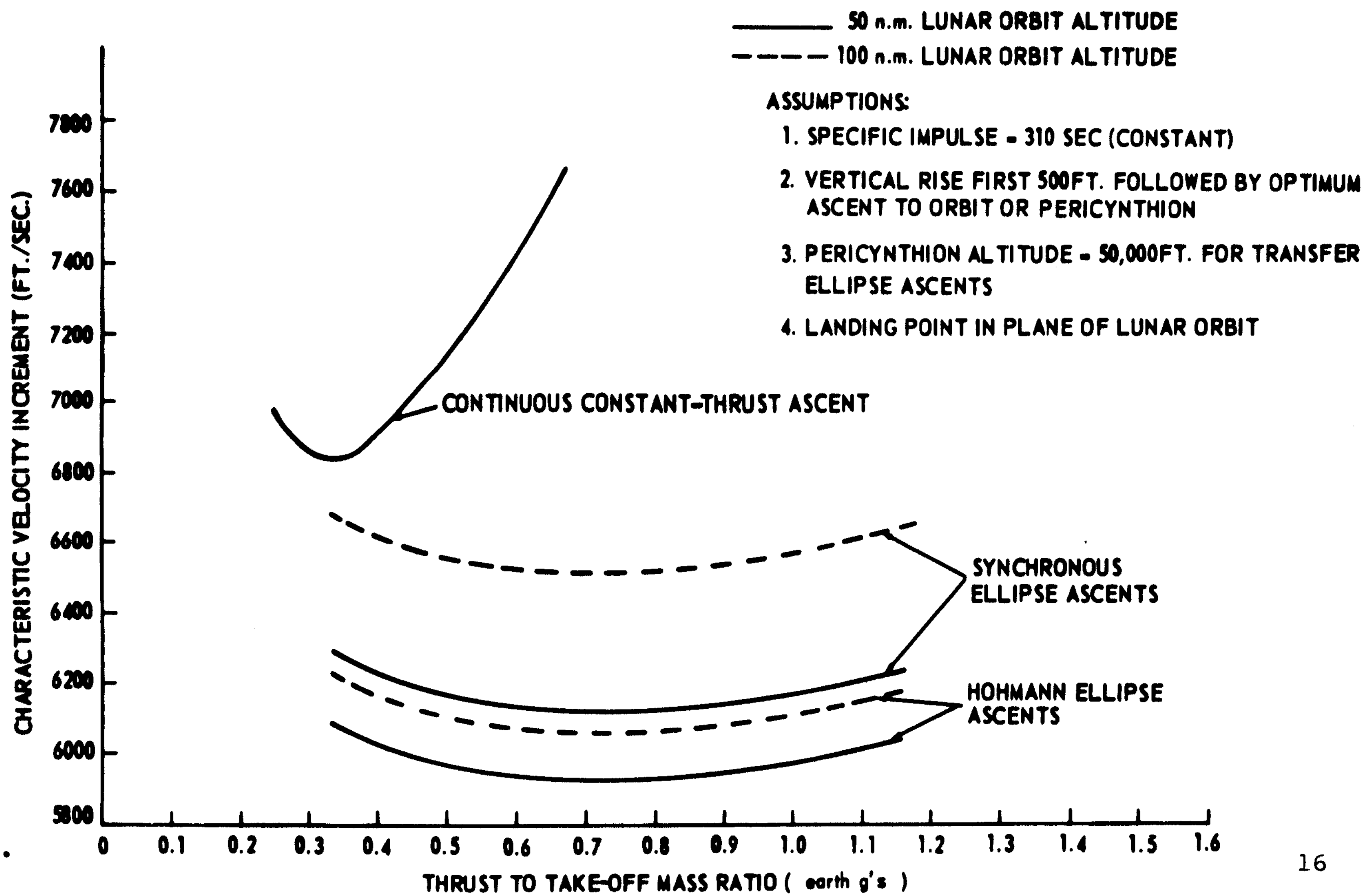
a. CONSTANT THRUST ASCENT



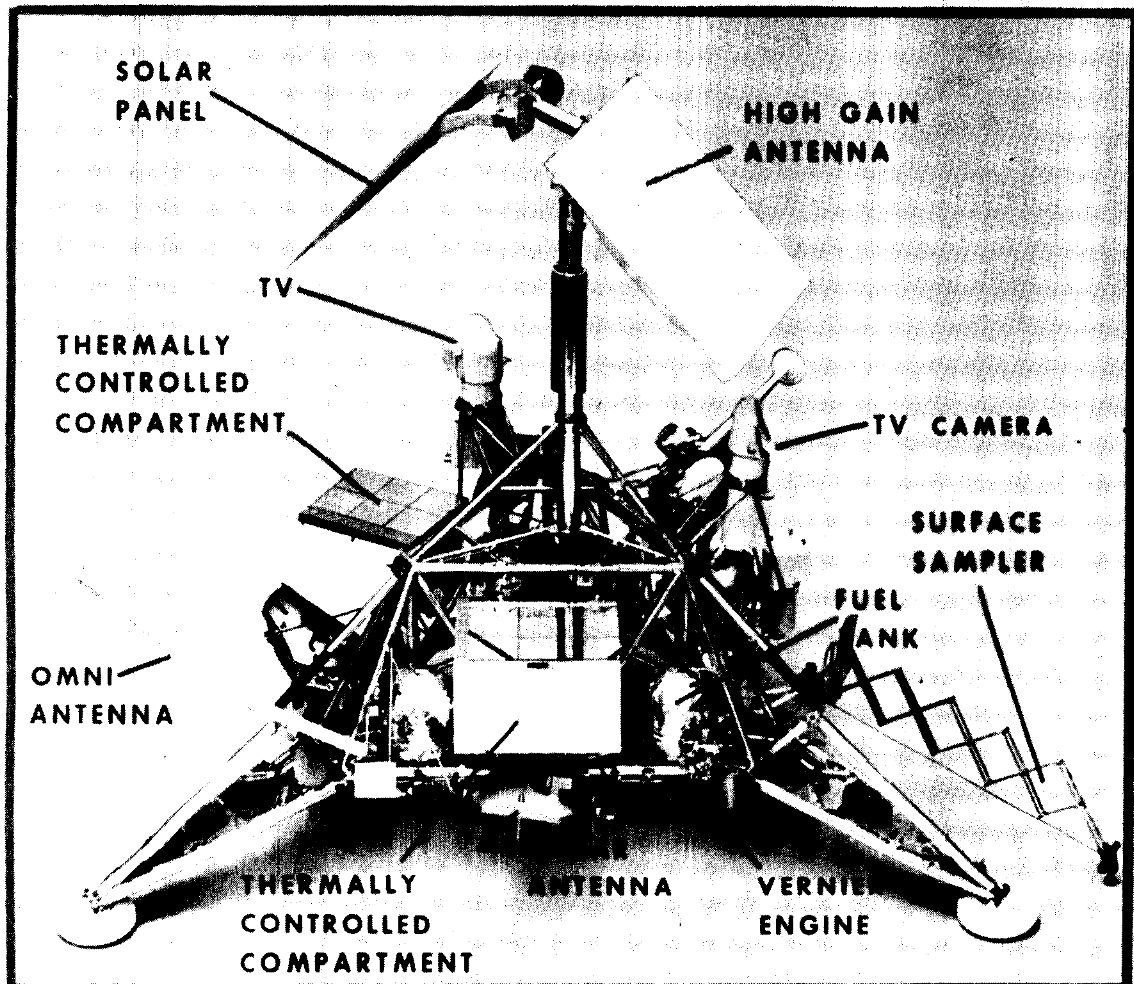
b. HOHMANN ELLIPSE ASCENT



c. SYNCHRONOUS ELLIPSE ASCENT


50 n.m. LUNAR ORBIT ALTITUDE
100 n.m. LUNAR ORBIT ALTITUDE
ASSUMPTIONS:
1. SPECIFIC IMPULSE = 310 SEC (CONSTANT)
2. VERTICAL RISE FIRST 500FT. FOLLOWED BY OPTIMUM ASCENT TO ORBIT OR PERICYNTHION
3. PERICYNTHION ALTITUDE = 50,000FT. FOR TRANSFER ELLIPSE ASCENTS
4. LANDING POINT IN PLANE OF LUNAR ORBIT
CONTINUOUS CONSTANT-THRUST ASCENT
SYNCHRONOUS ELLIPSE ASCENTS
HOHMANN ELLIPSE ASCENTS
CHARACTERISTIC VELOCITY INCREMENT (FT./SEC.)
7800
7600
7400
7200
7000
6800
6600
6400
6200
6000
5800
0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0.9
1.0
1.1
1.2
1.3
1.4
1.5
1.6
THRUST TO TAKE-OFF MASS RATIO (earth g's)

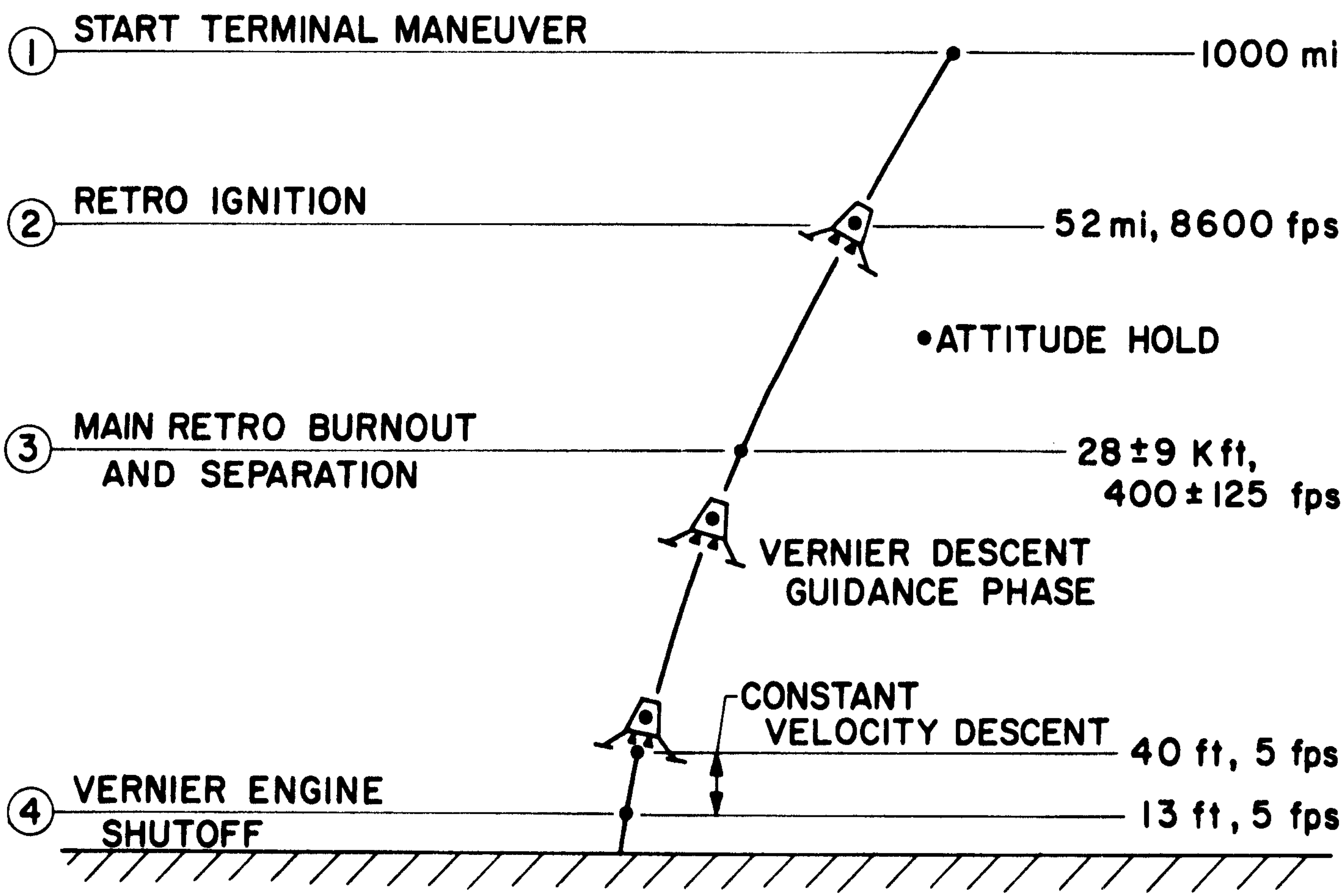
SURVEYOR SPACECRAFT



- **DEMONSTRATE**
  SOFT LANDING TECHNOLOGY
- **SURVEY**
  VARIOUS LANDING AREAS
- **MEASURE**
  PHYSICAL & CHEMICAL PROPERTIES

NASA S63-261

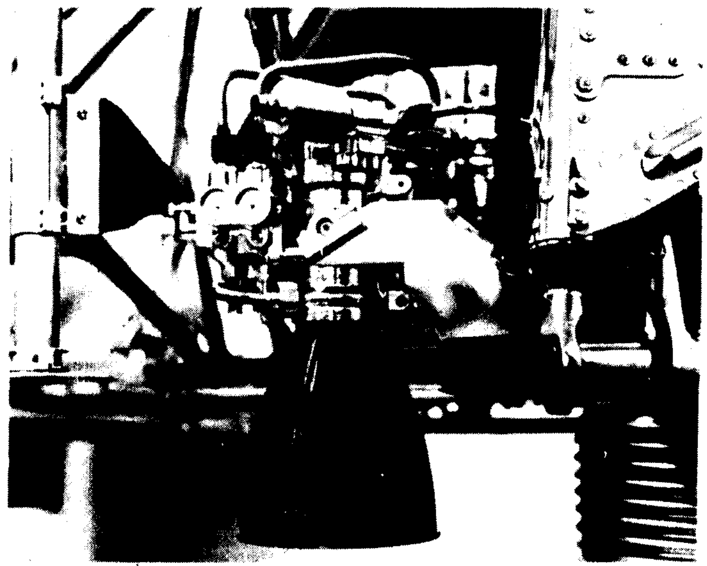
SURVEYOR TERMINAL DESCENT PHASE



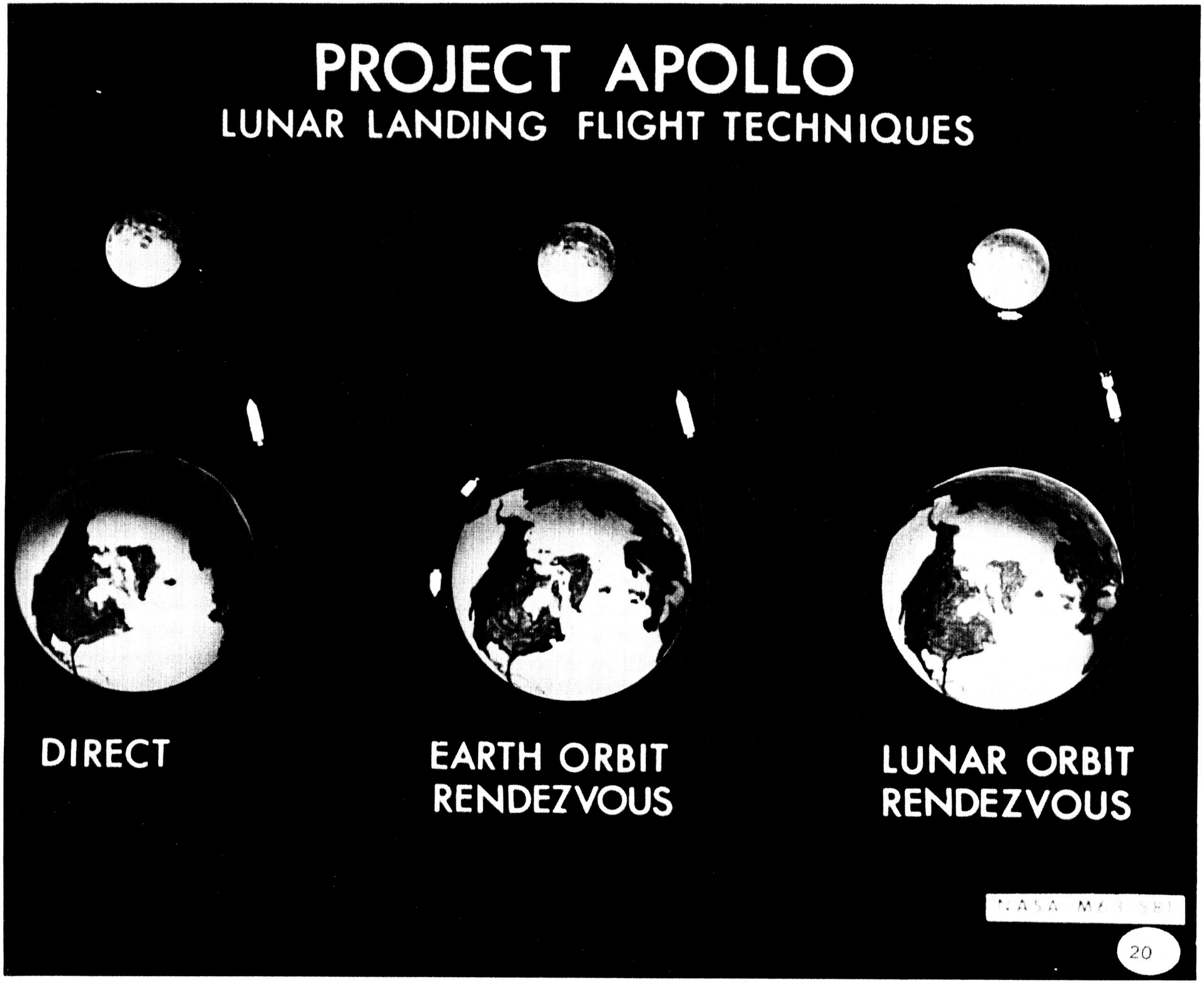
NASA SL64-281

SURVEYOR LANDING SUBSYSTEMS

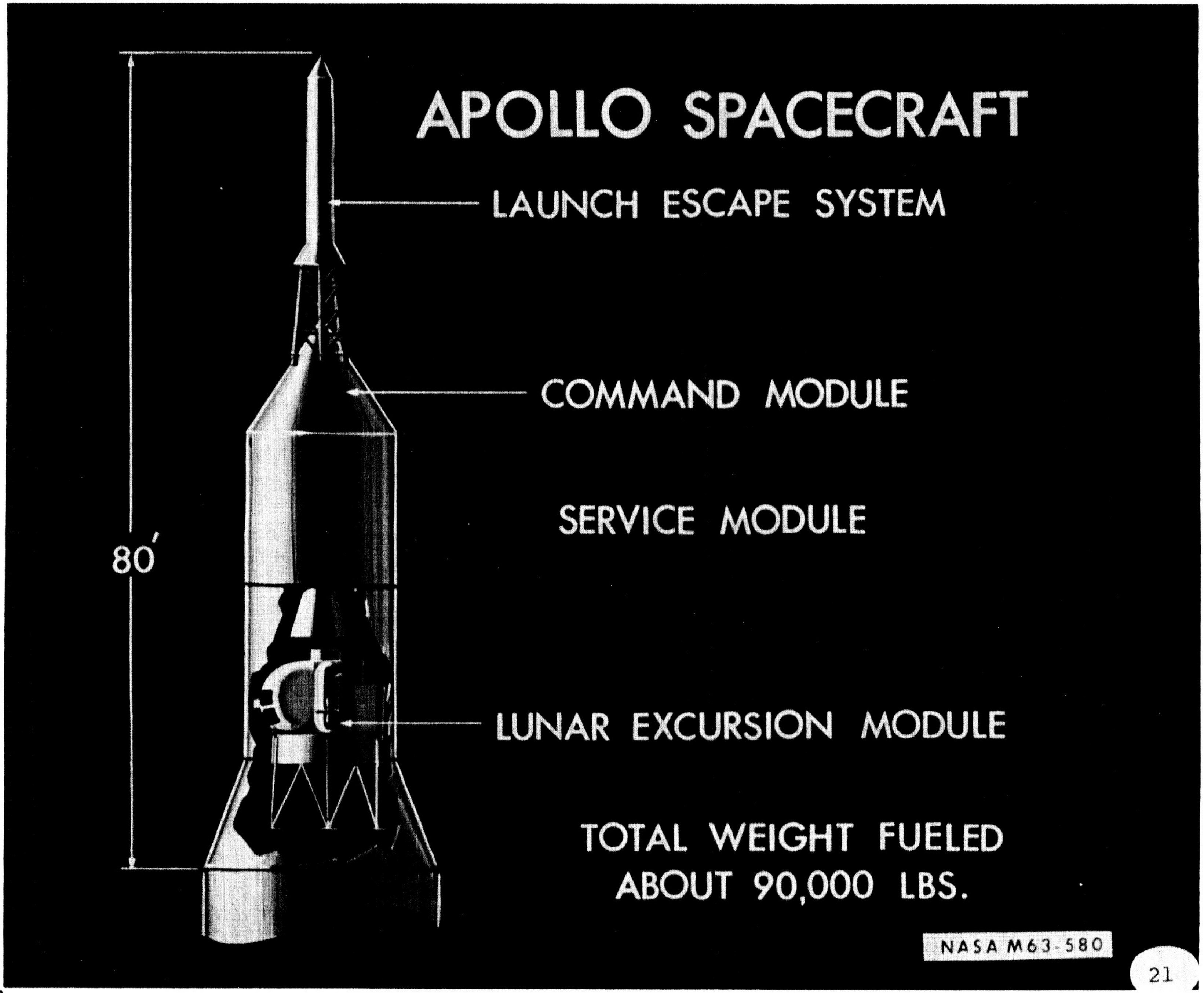
MAIN
RETROROCKET



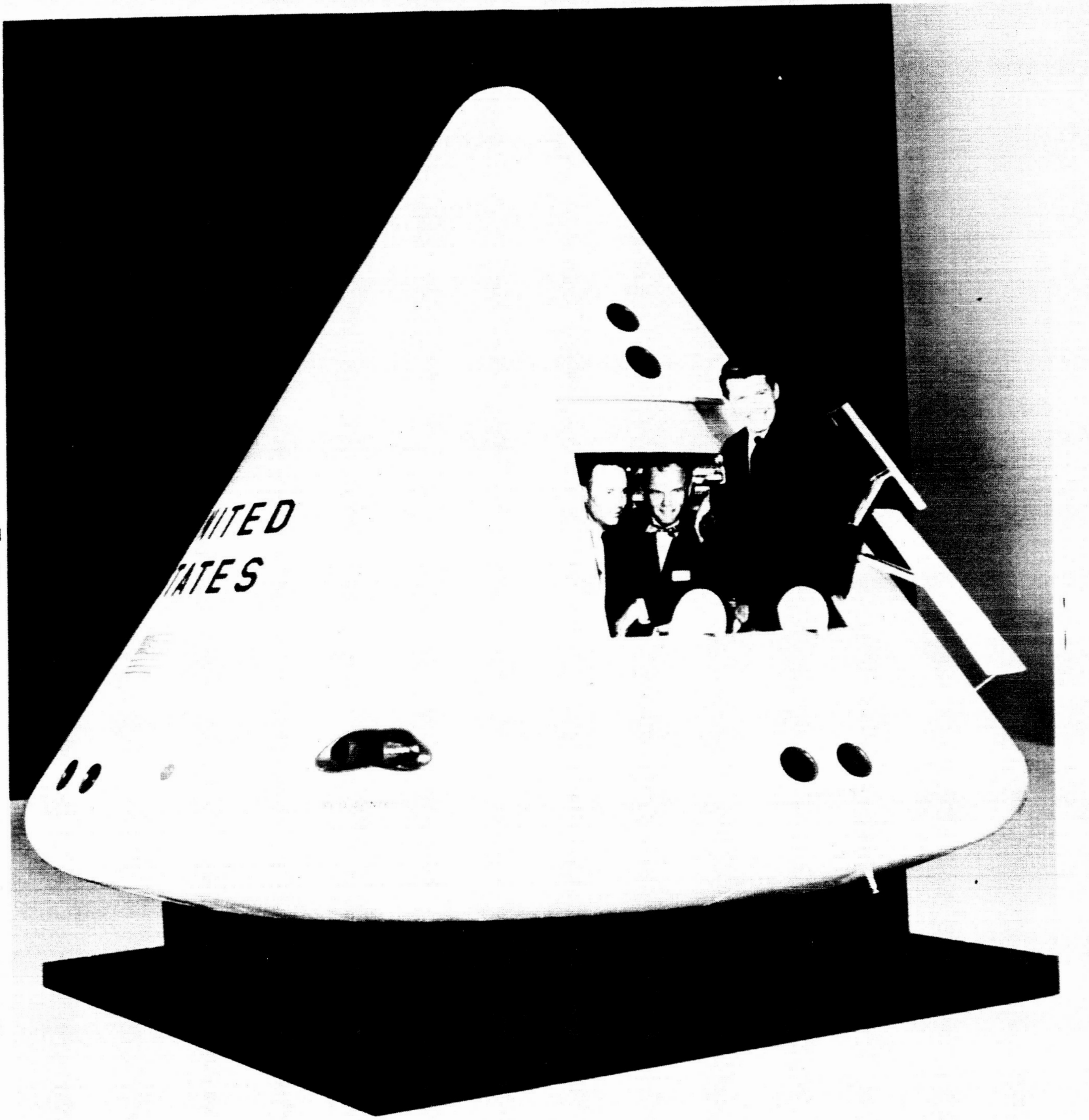
VERNIER ENGINE



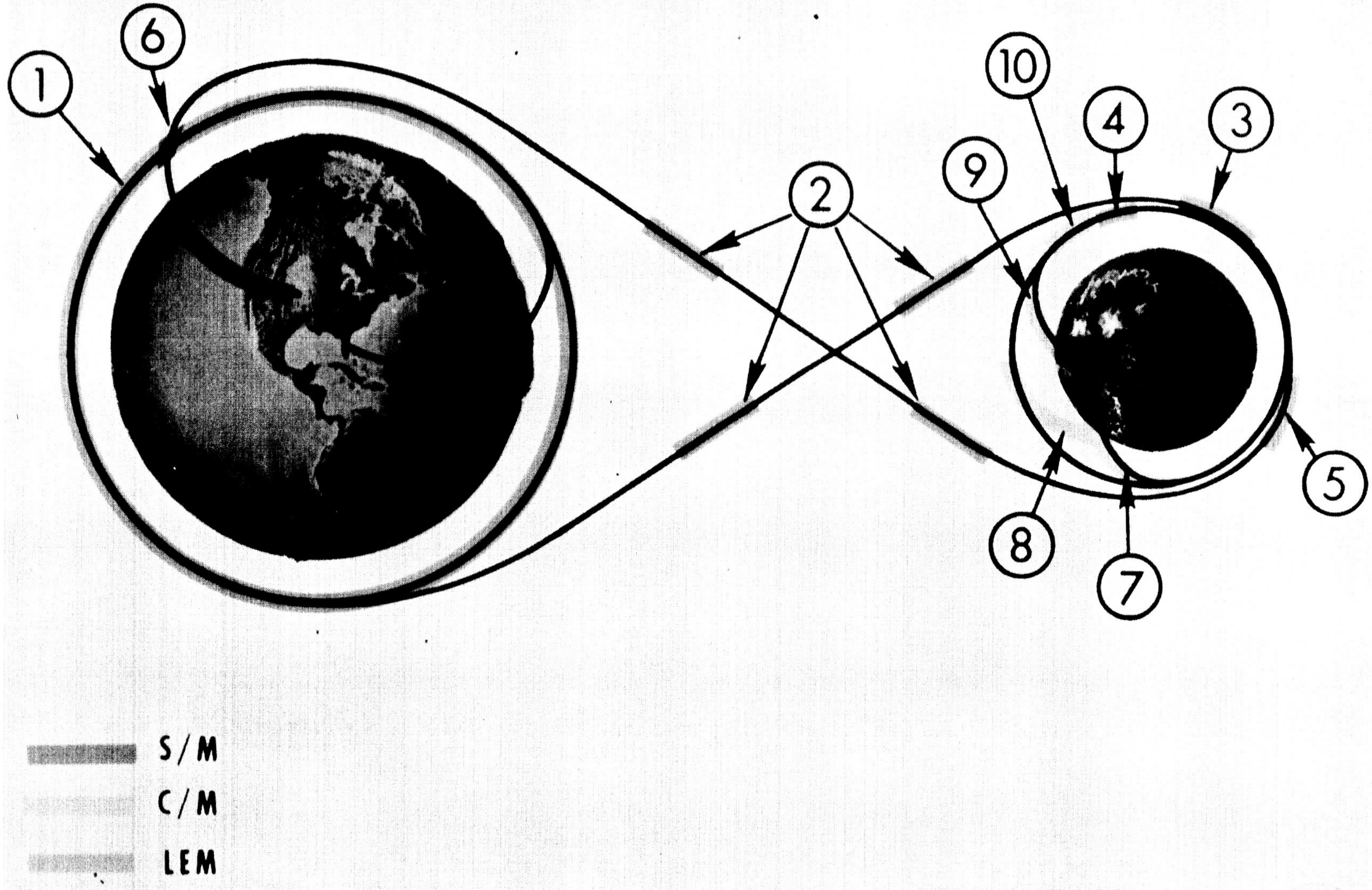
PROJECT APOLLO
LUNAR LANDING FLIGHT TECHNIQUES
DIRECT
EARTH ORBIT RENDEZVOUS
LUNAR ORBIT RENDEZVOUS
20


APOLLO SPACECRAFT
LAUNCH ESCAPE SYSTEM
COMMAND MODULE
SERVICE MODULE
LUNAR EXCURSION MODULE
TOTAL WEIGHT FUELED
ABOUT 90,000 LBS.
80′
NASA M63-580

COMMAND MODULE



MANNED LUNAR LANDING
LUNAR ORBIT RENDEZVOUS

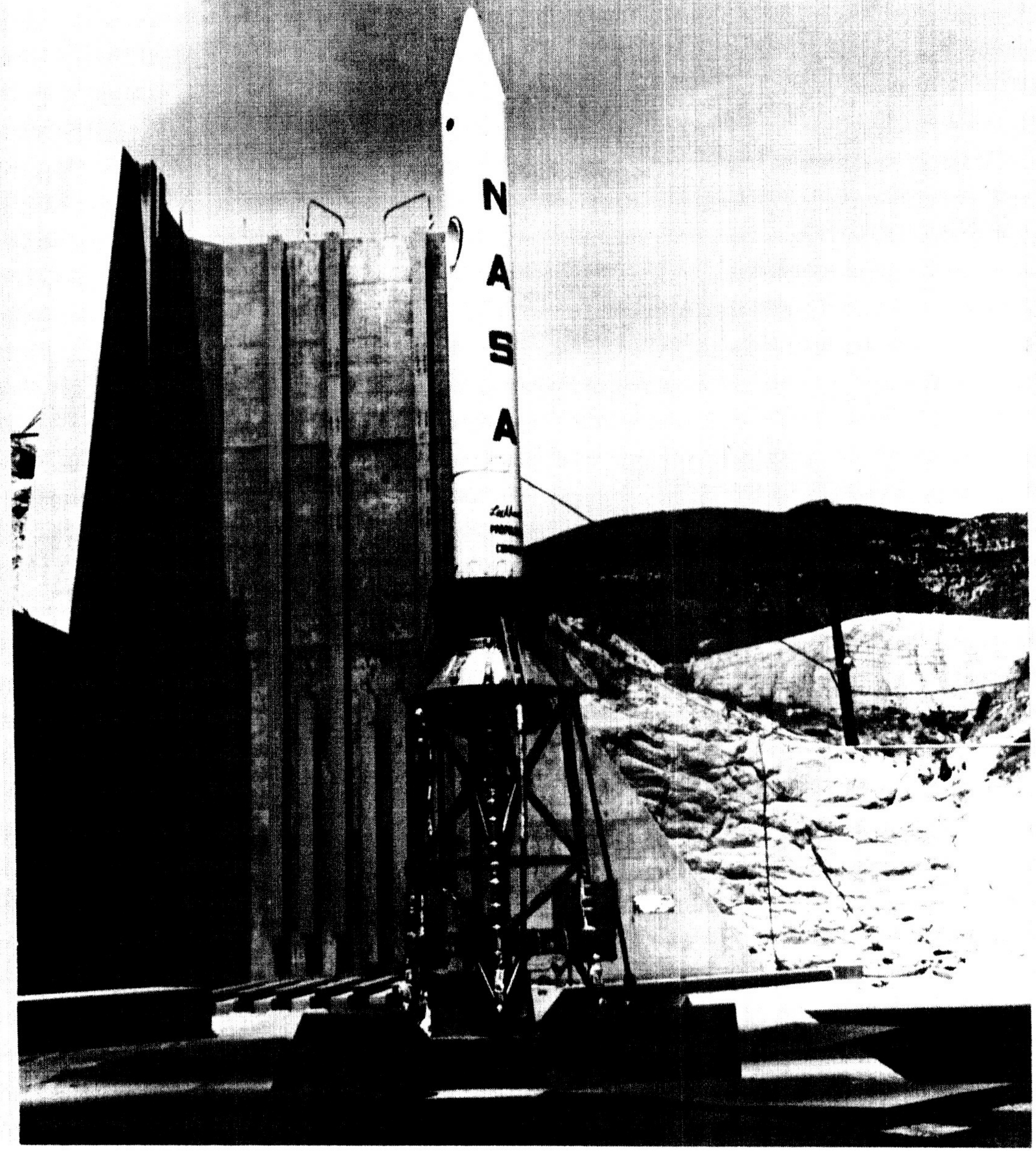


APOLLO

ON BOARD PROPULSION REQUIREMENTS

[redacted]

CM	6	**RE-ENTRY CONTROL**
LEM	7	**DESCENT FROM LUNAR ORBIT**
	8	**ABORT**
	9	**ASCENT INTO LUNAR ORBIT**
	10	**RENDEZVOUS**


NASA

COMPARISON OF PROPELLANT LIQUID RANGE AND PLANET SURFACE TEMPERATURES

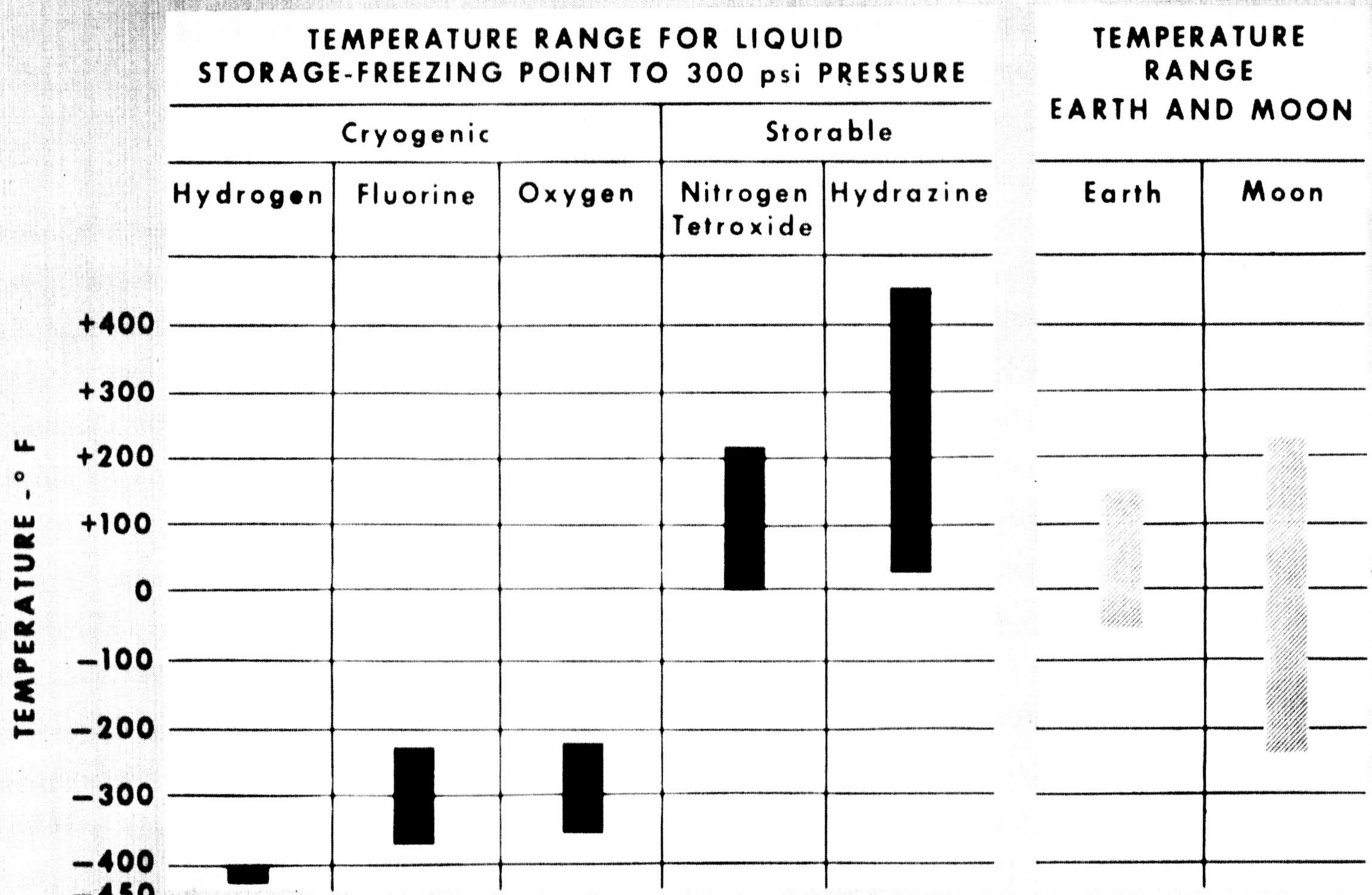
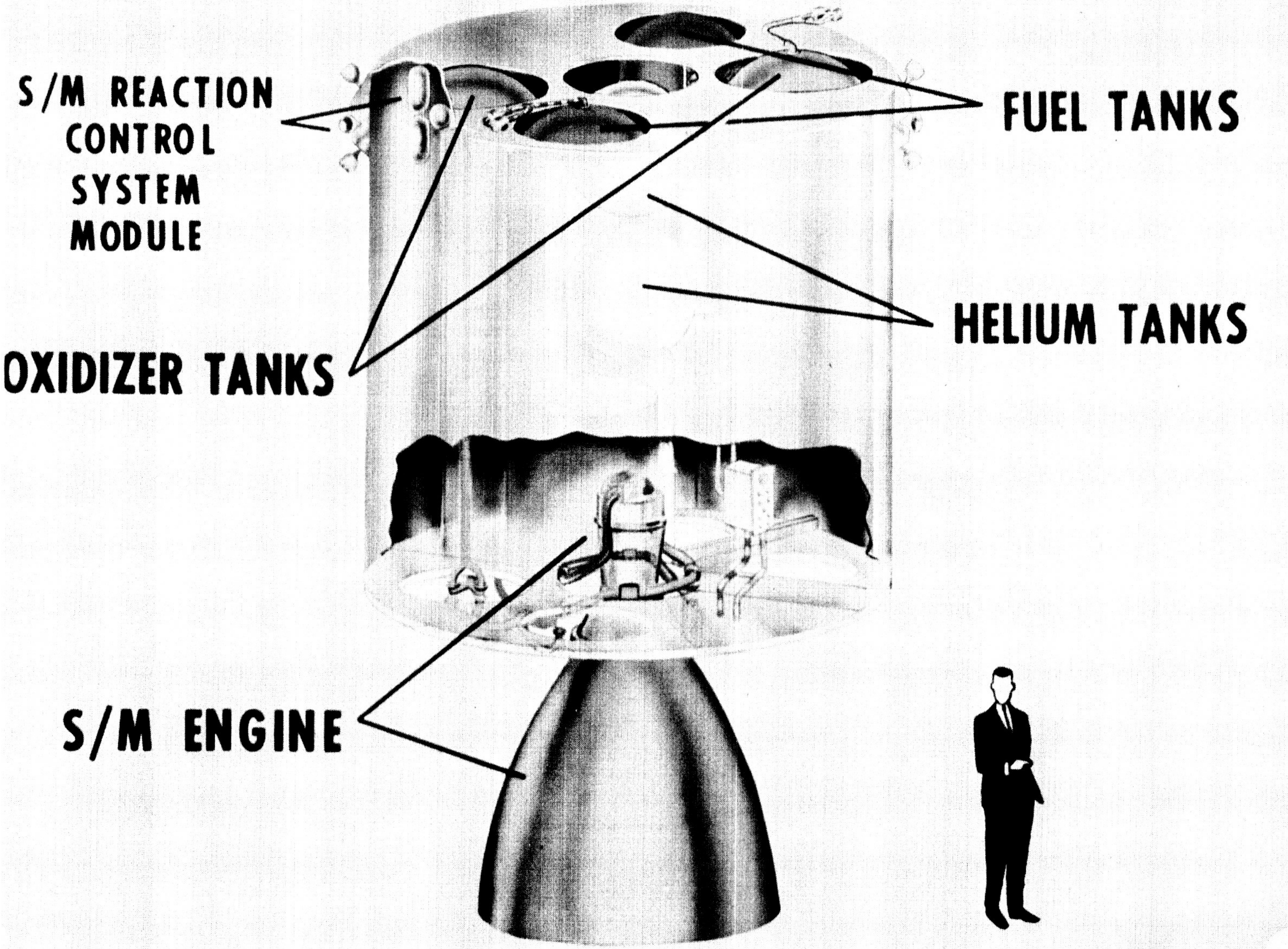


FIGURE 17

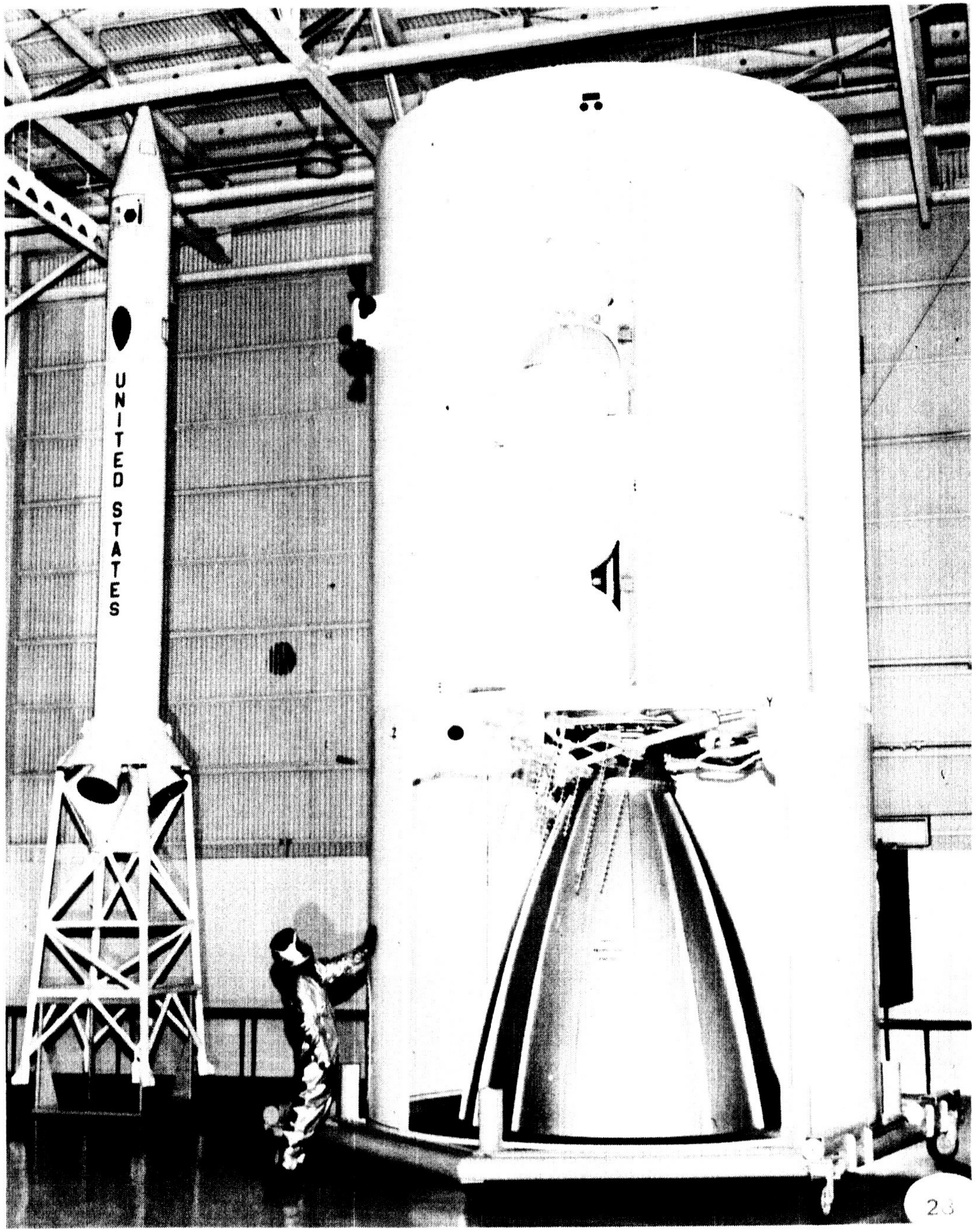
APOLLO SERVICE MODULE PROPULSION SYSTEMS



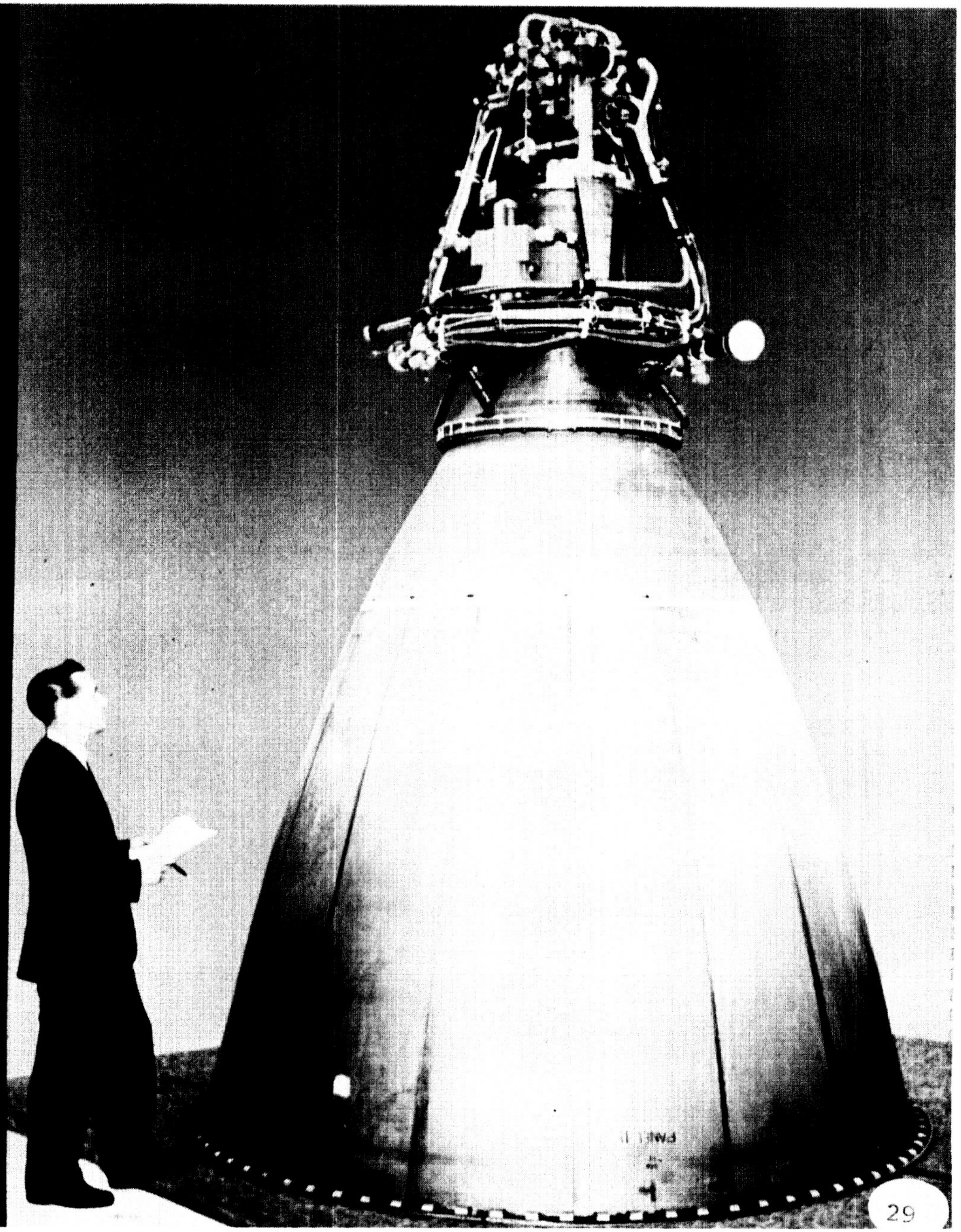
SERVICE MODULE

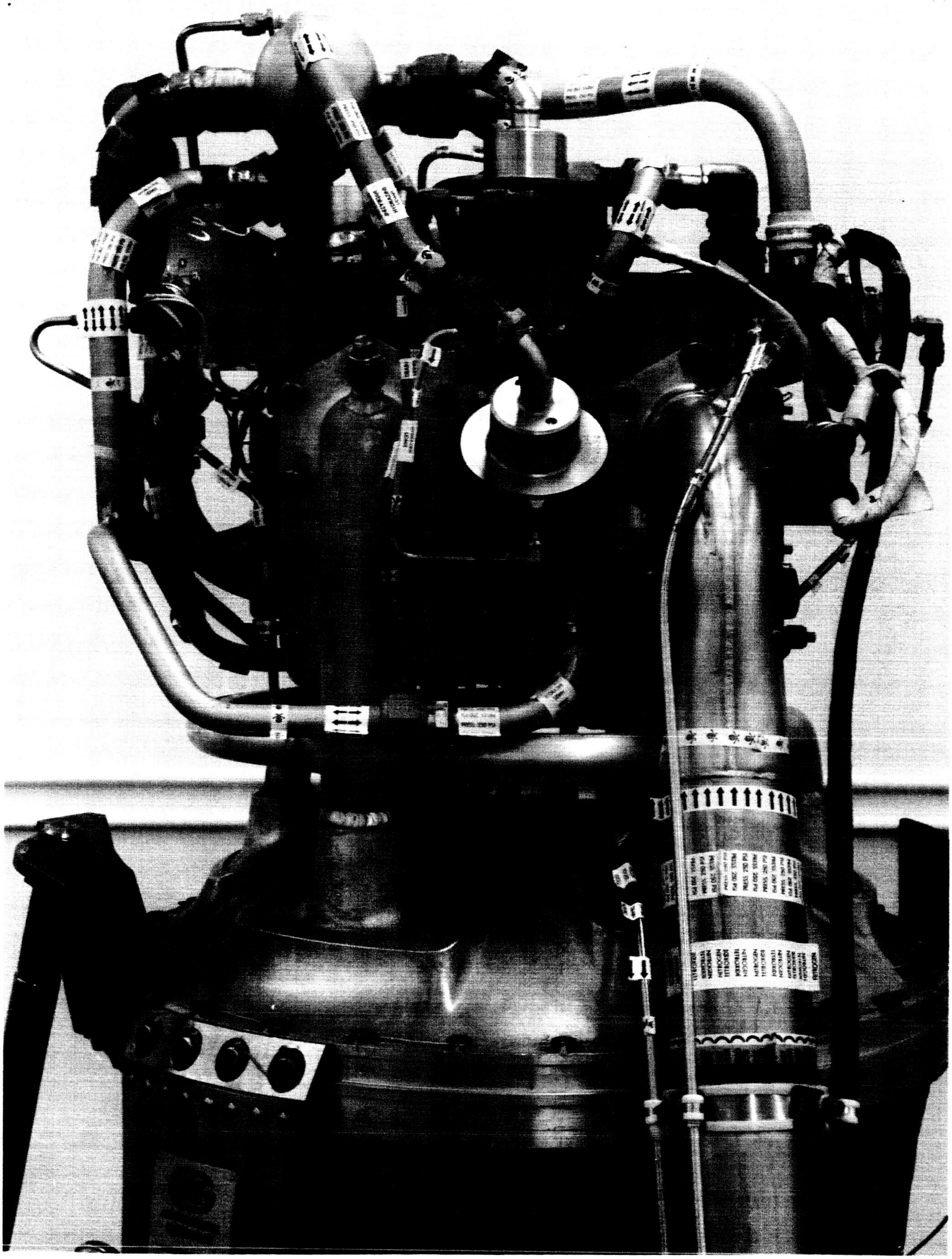
AND

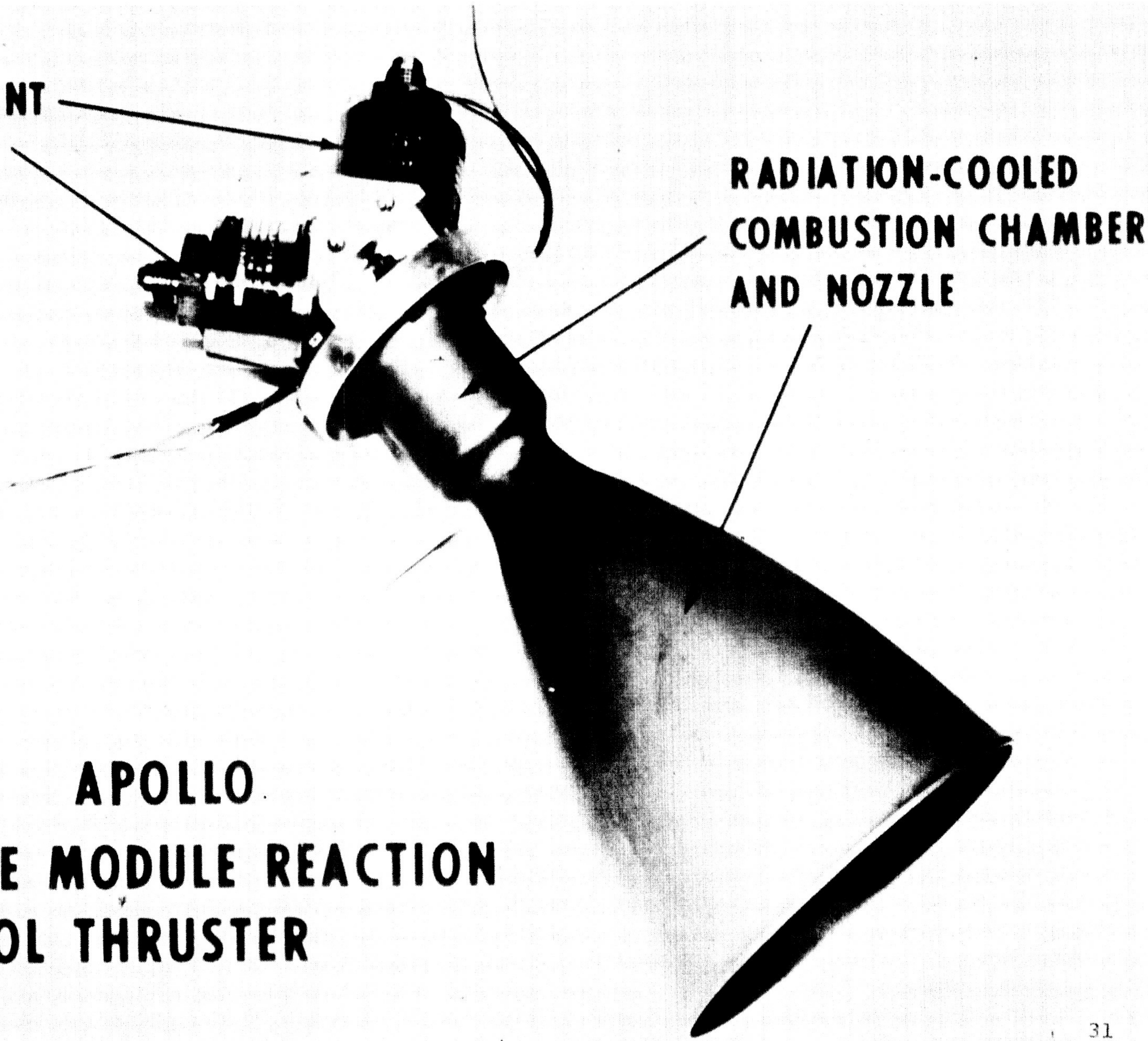
ESCAPE TOWER



APOLLO
SERVICE MODULE ENGINE





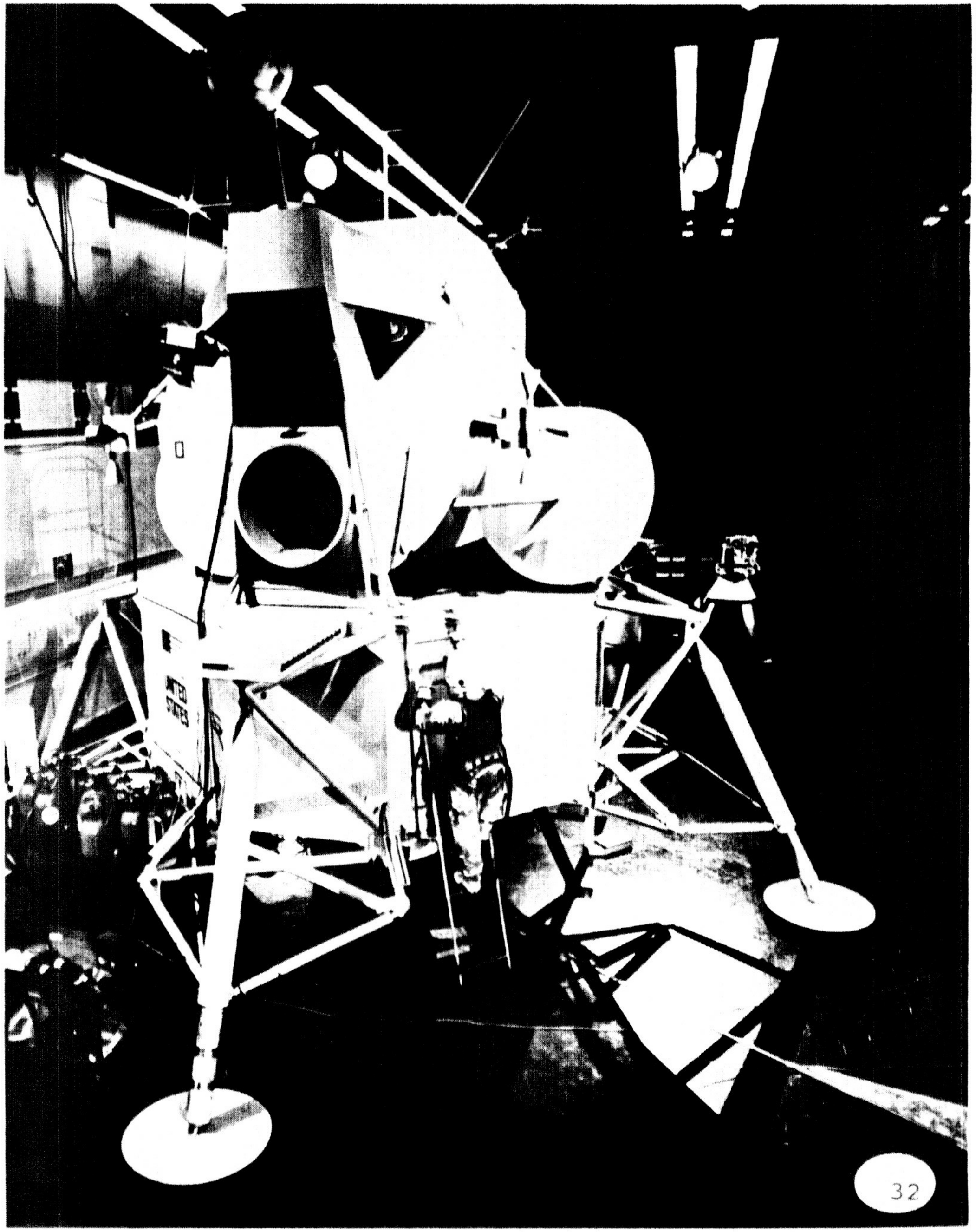


APOLLO
SERVICE MODULE REACTION
CONTROL THRUSTER

LUNAR

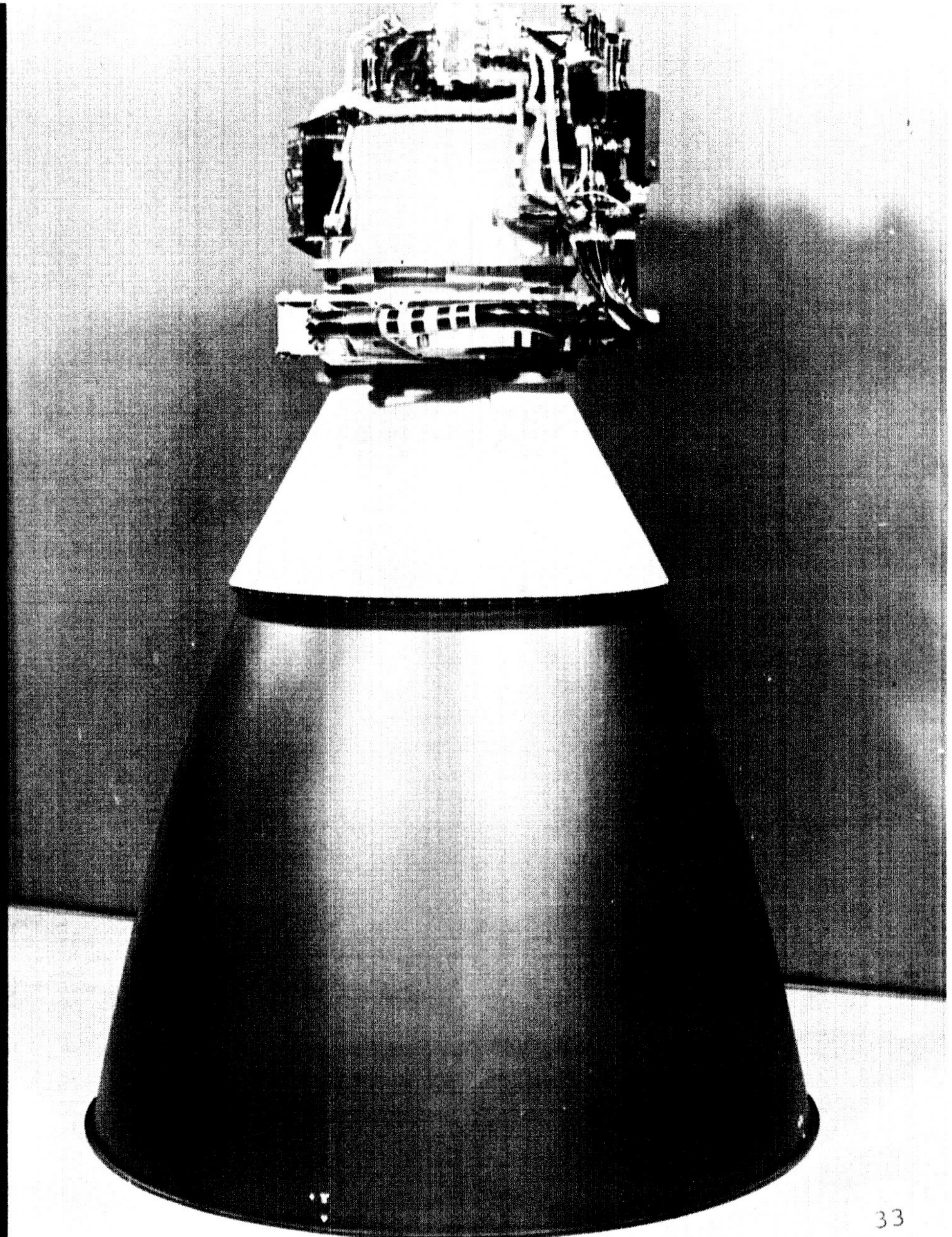
EXCURSION

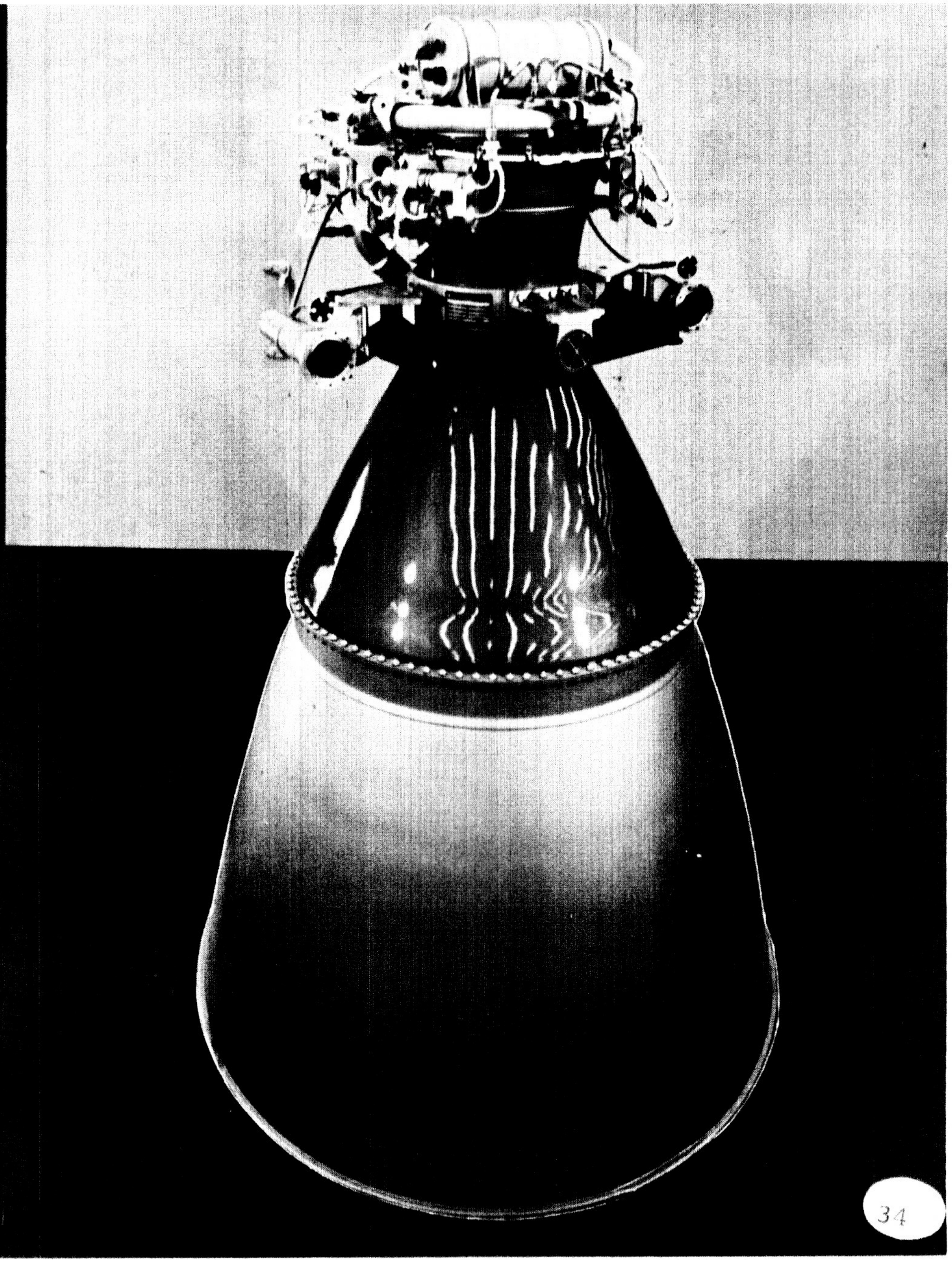
MODULE



APOLLO

LUNAR EXCURSION MODULE
DESCENT ENGINE





LUNAR EXCURSION MODULE DESCENT ENGINE (ROCKETDYNE)

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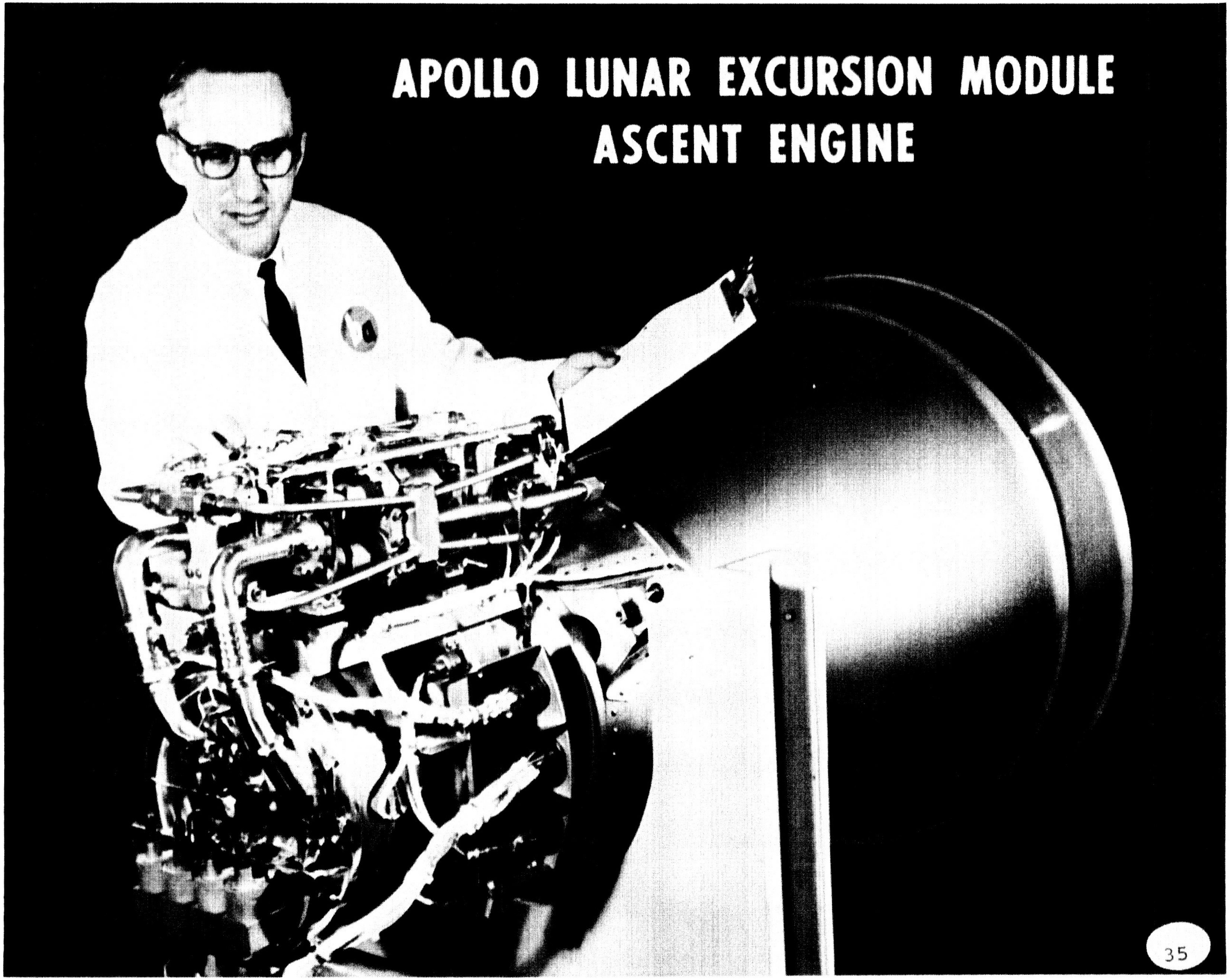
APOLLO LUNAR EXCURSION MODULE
ASCENT ENGINE
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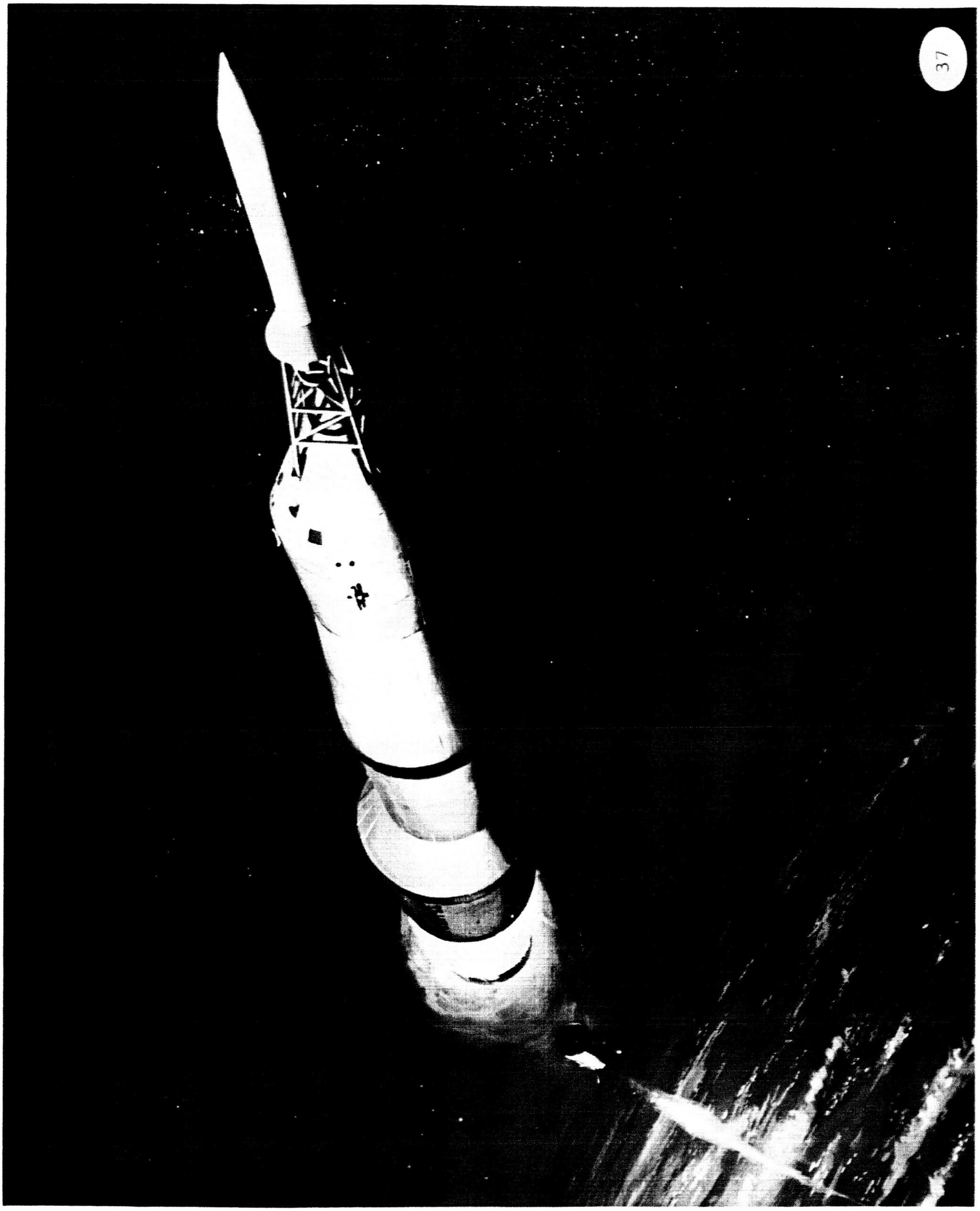

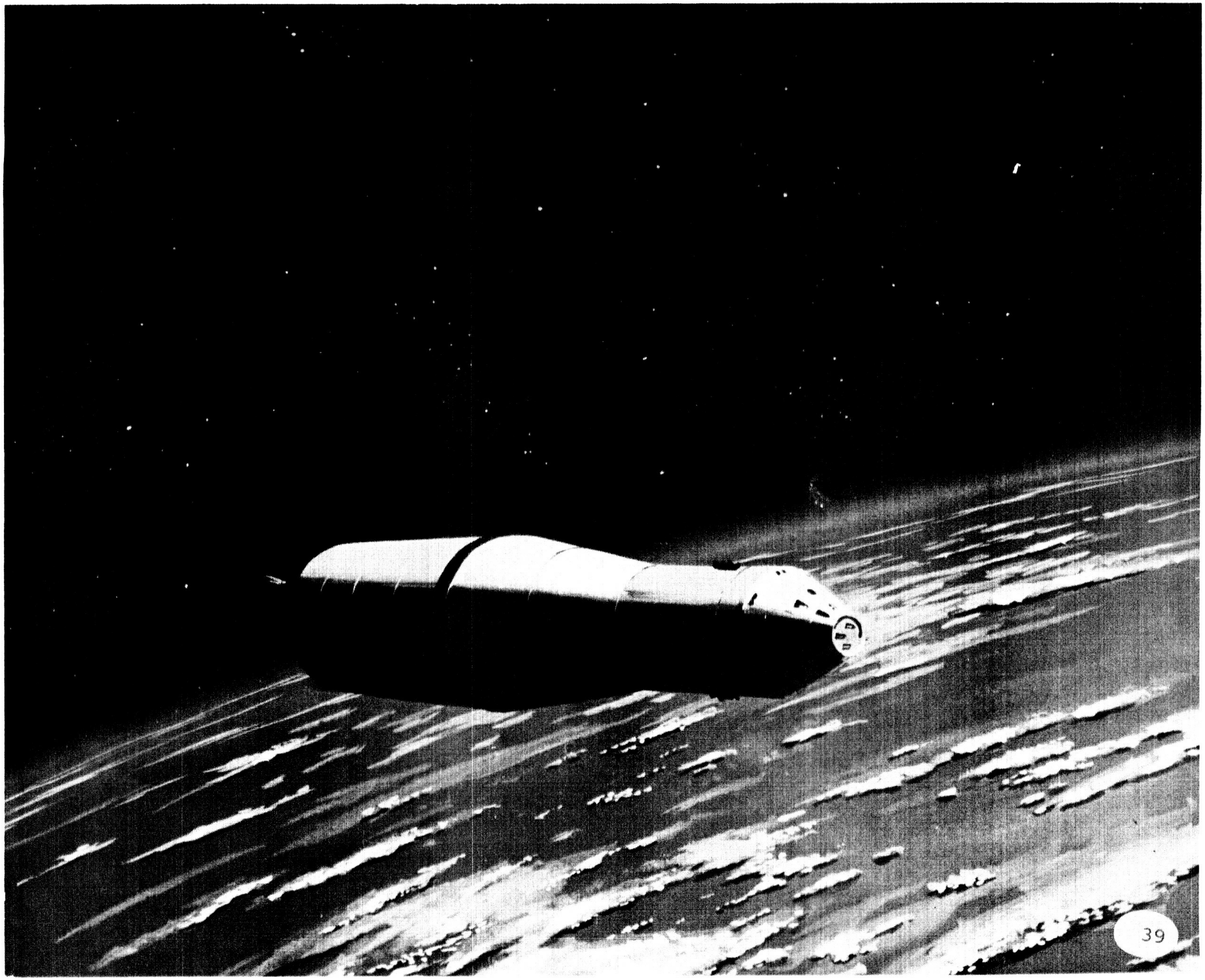

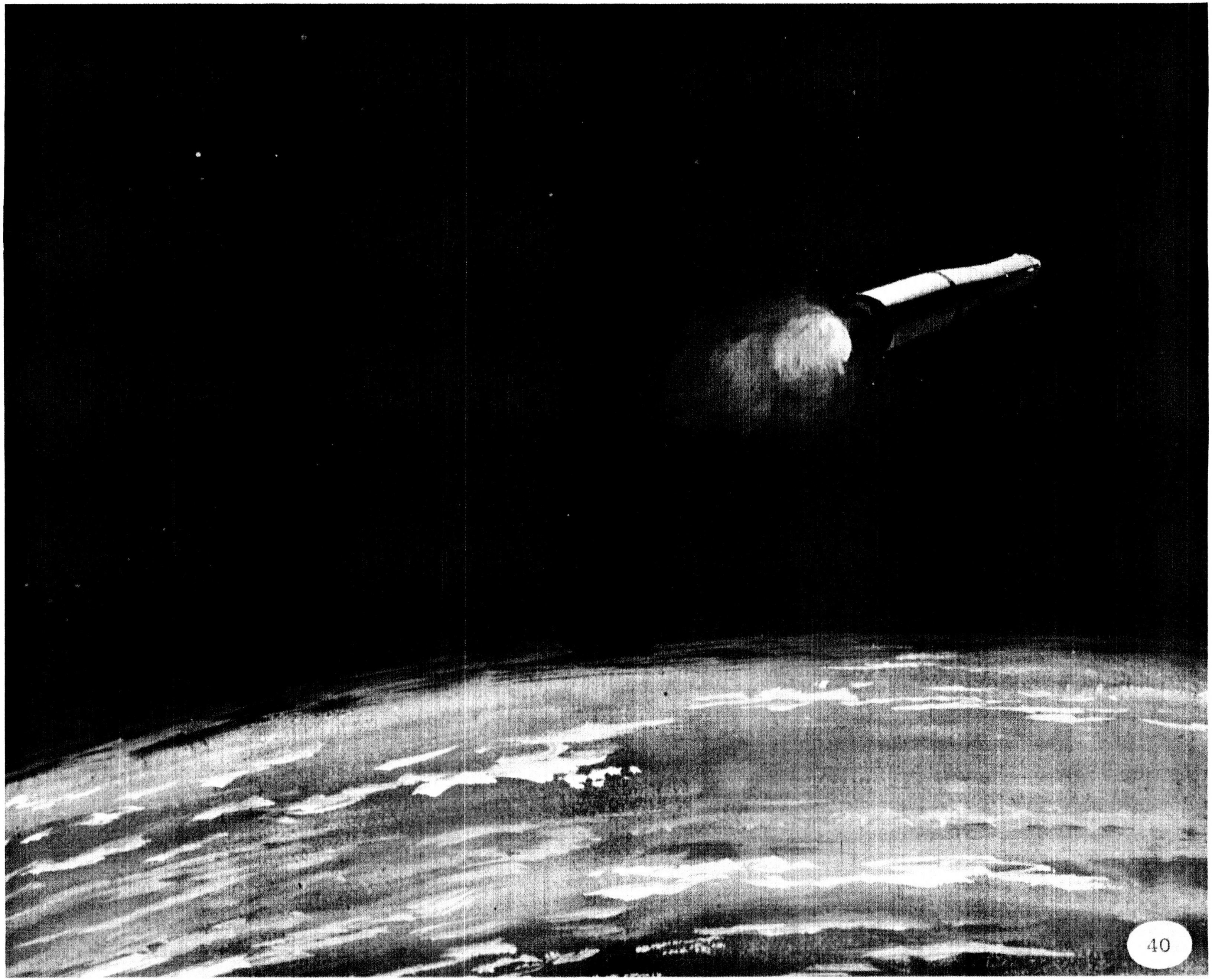
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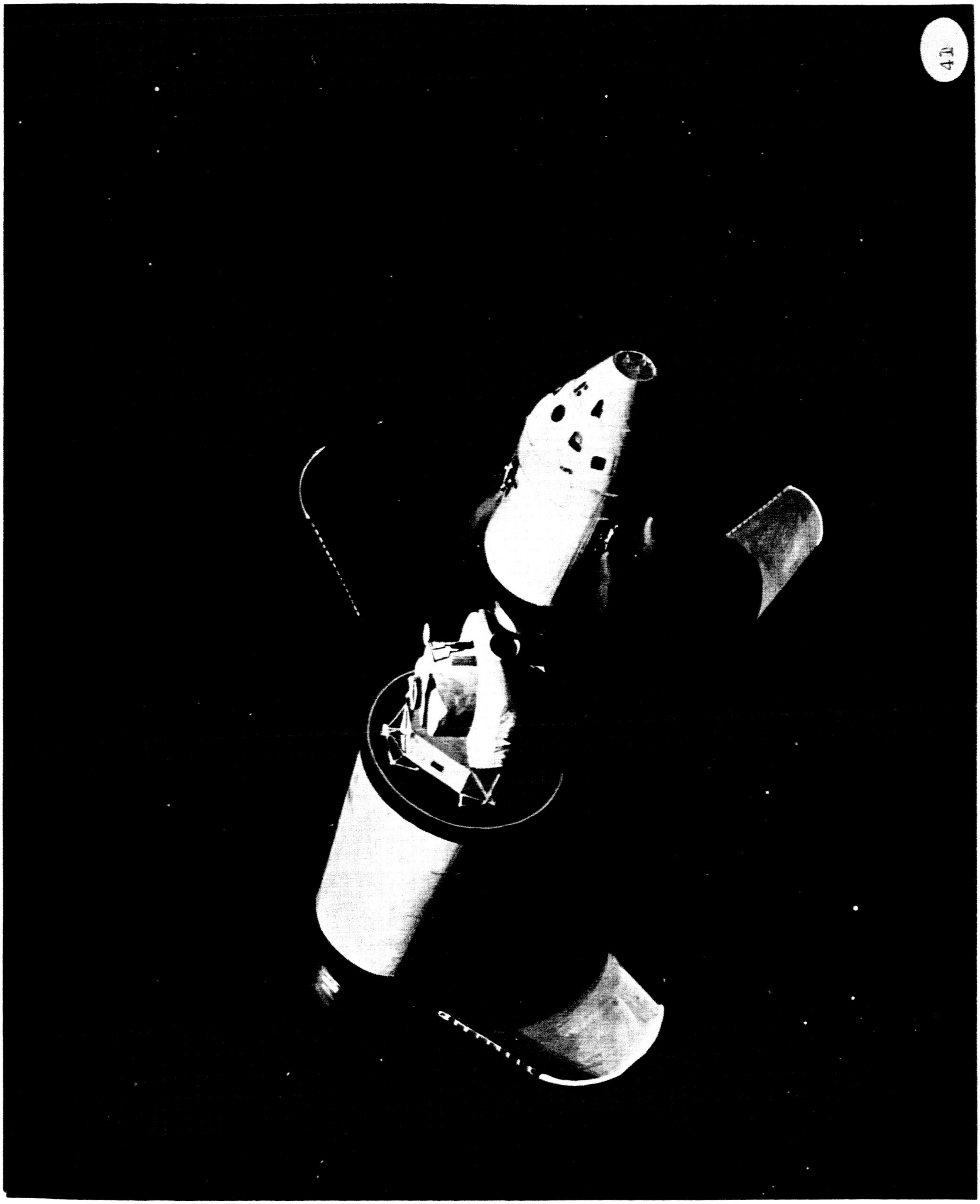
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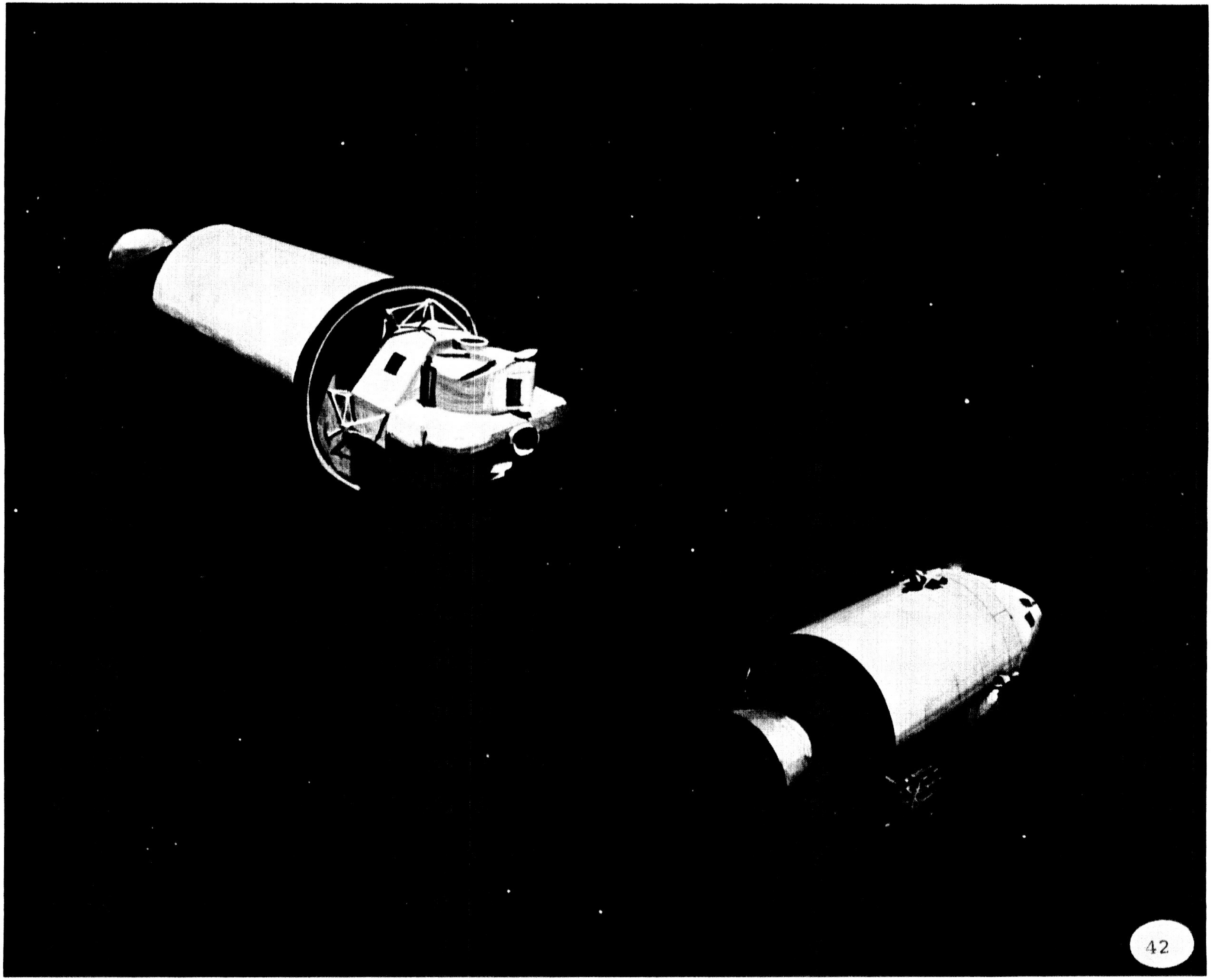
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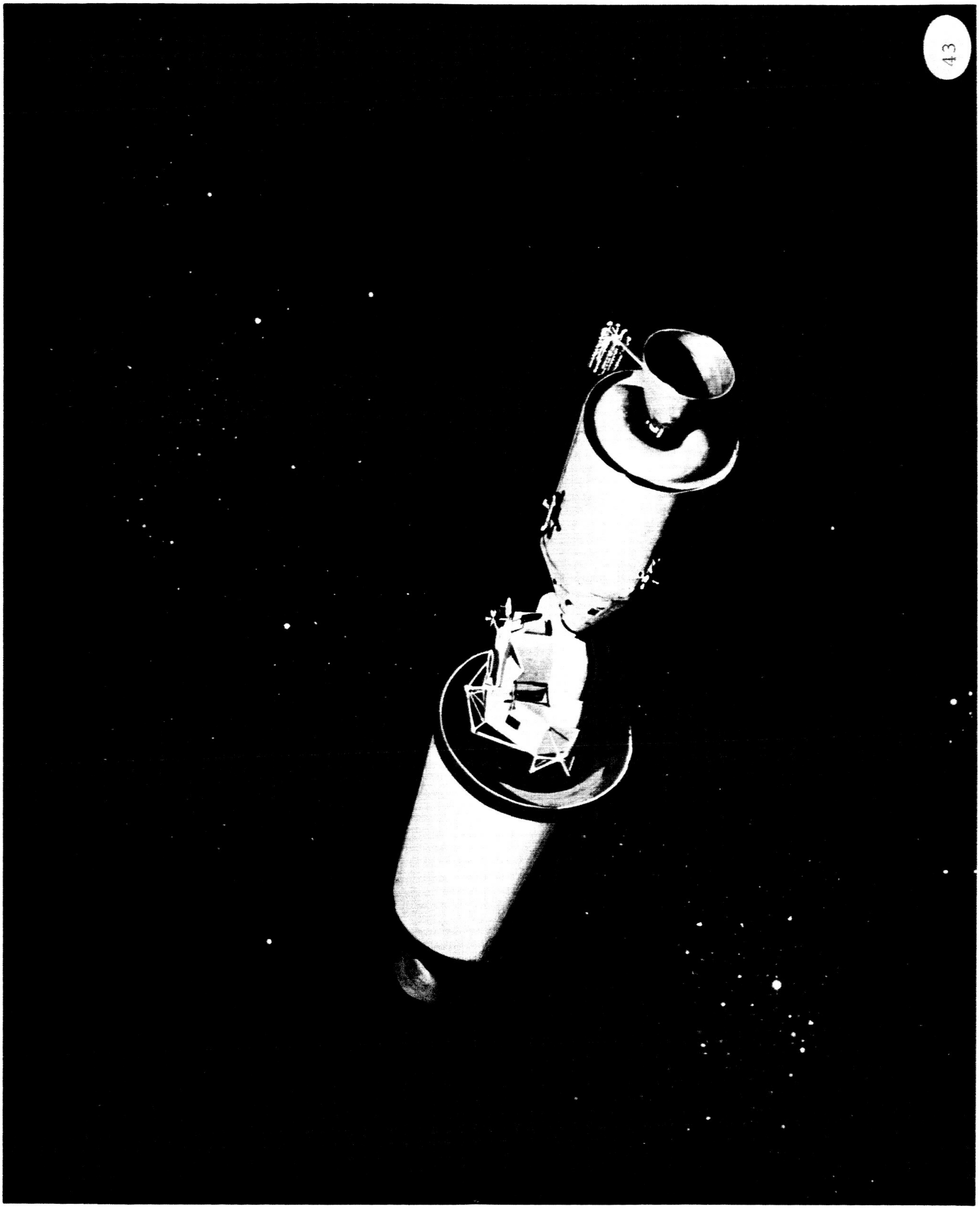

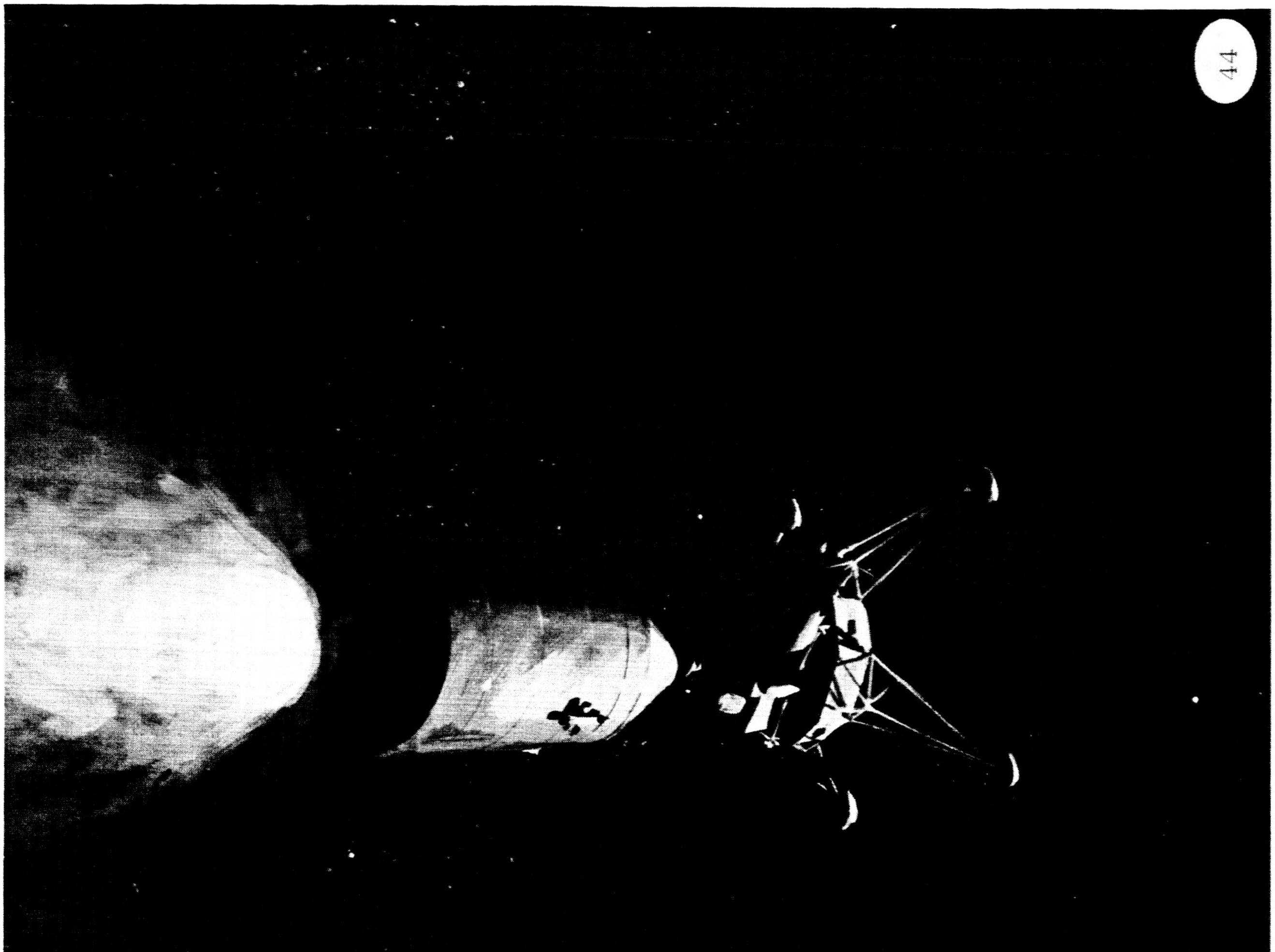
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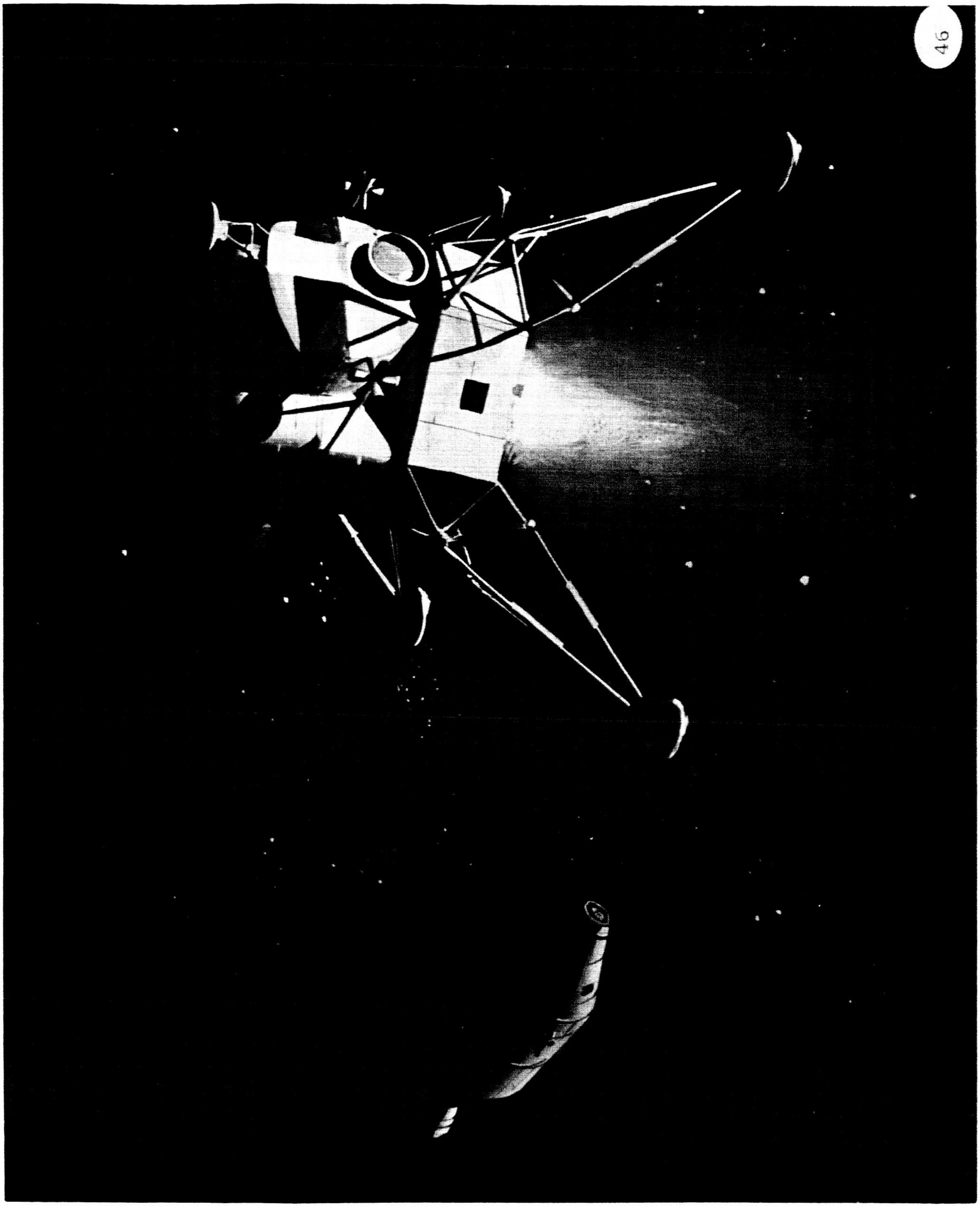

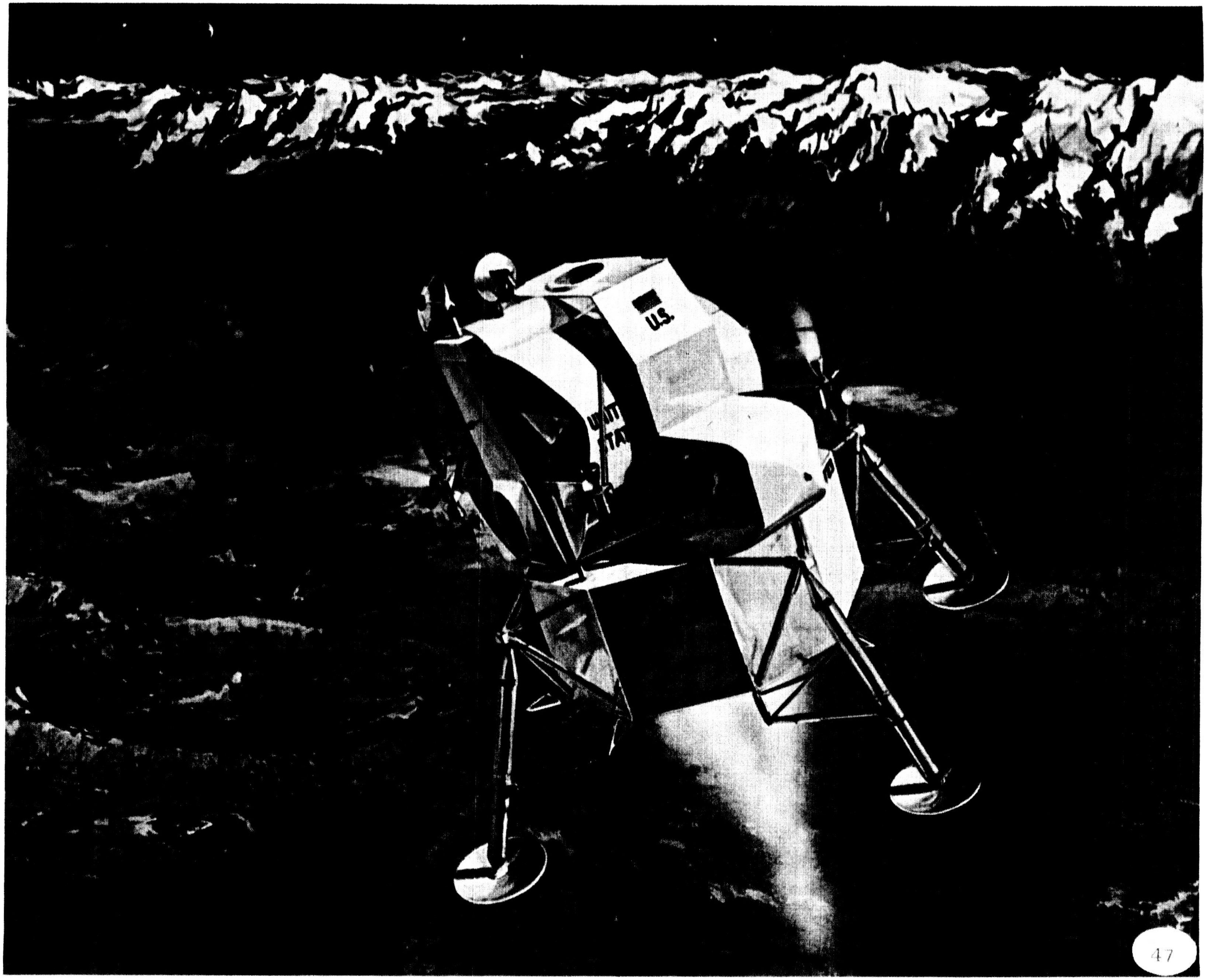
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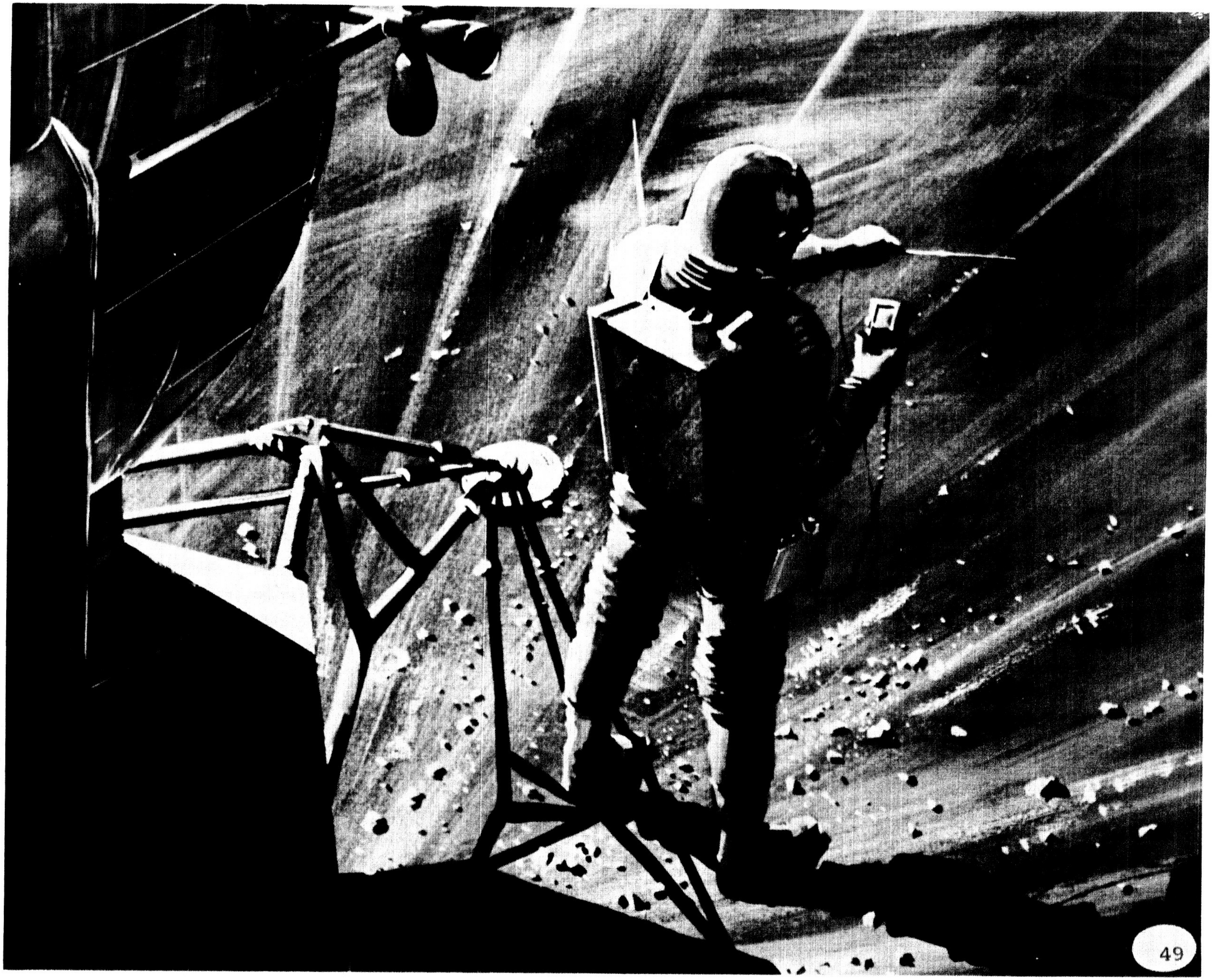
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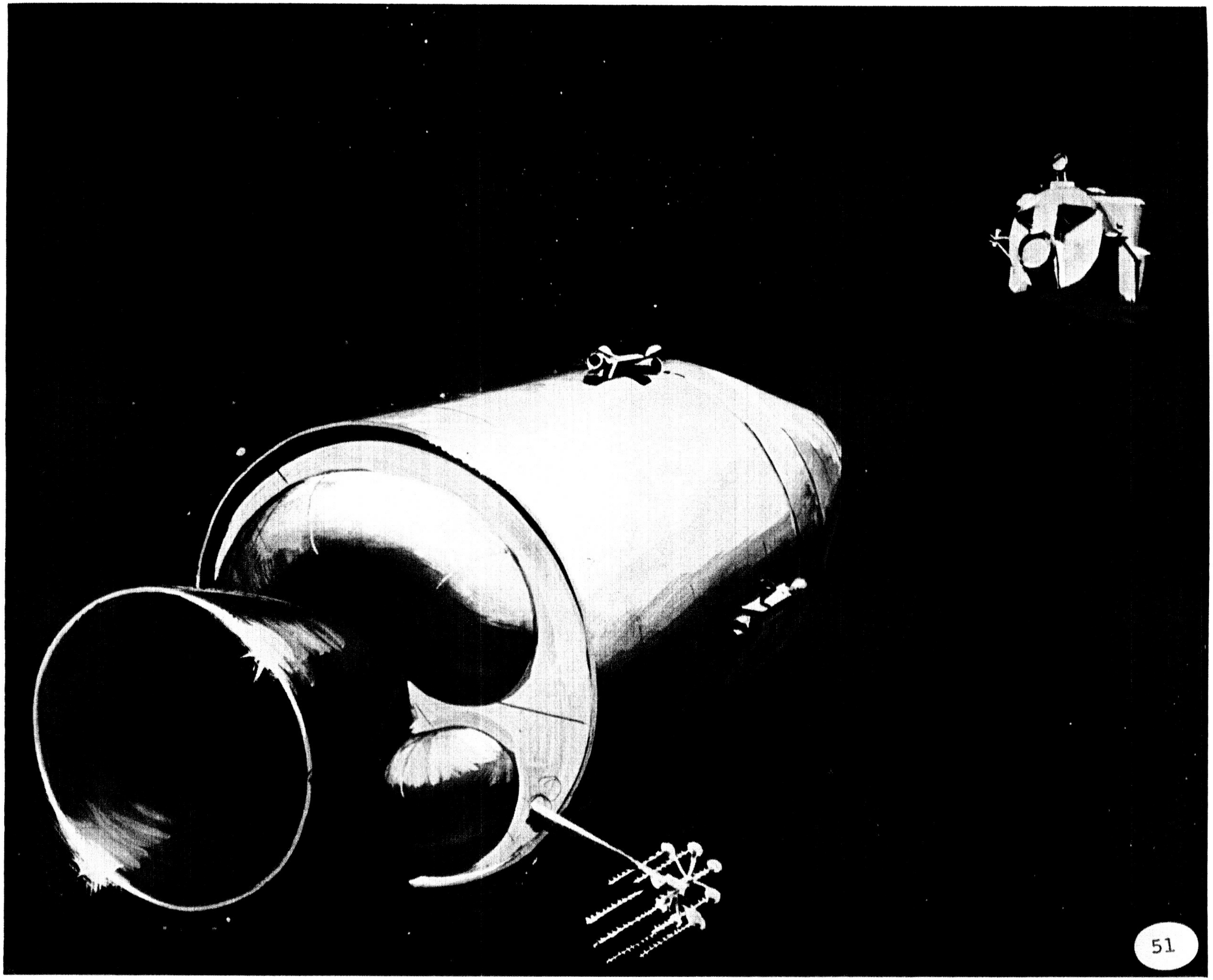

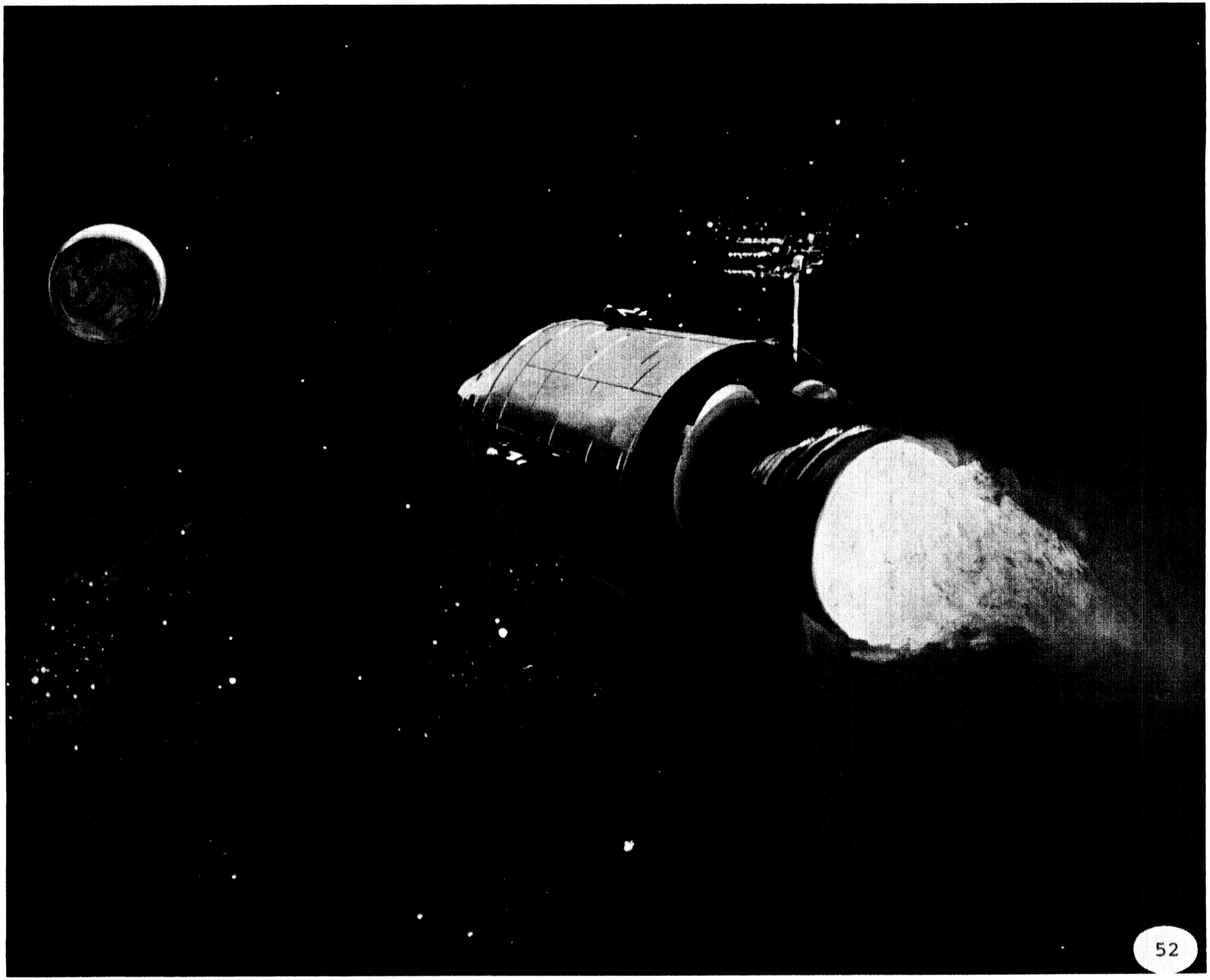
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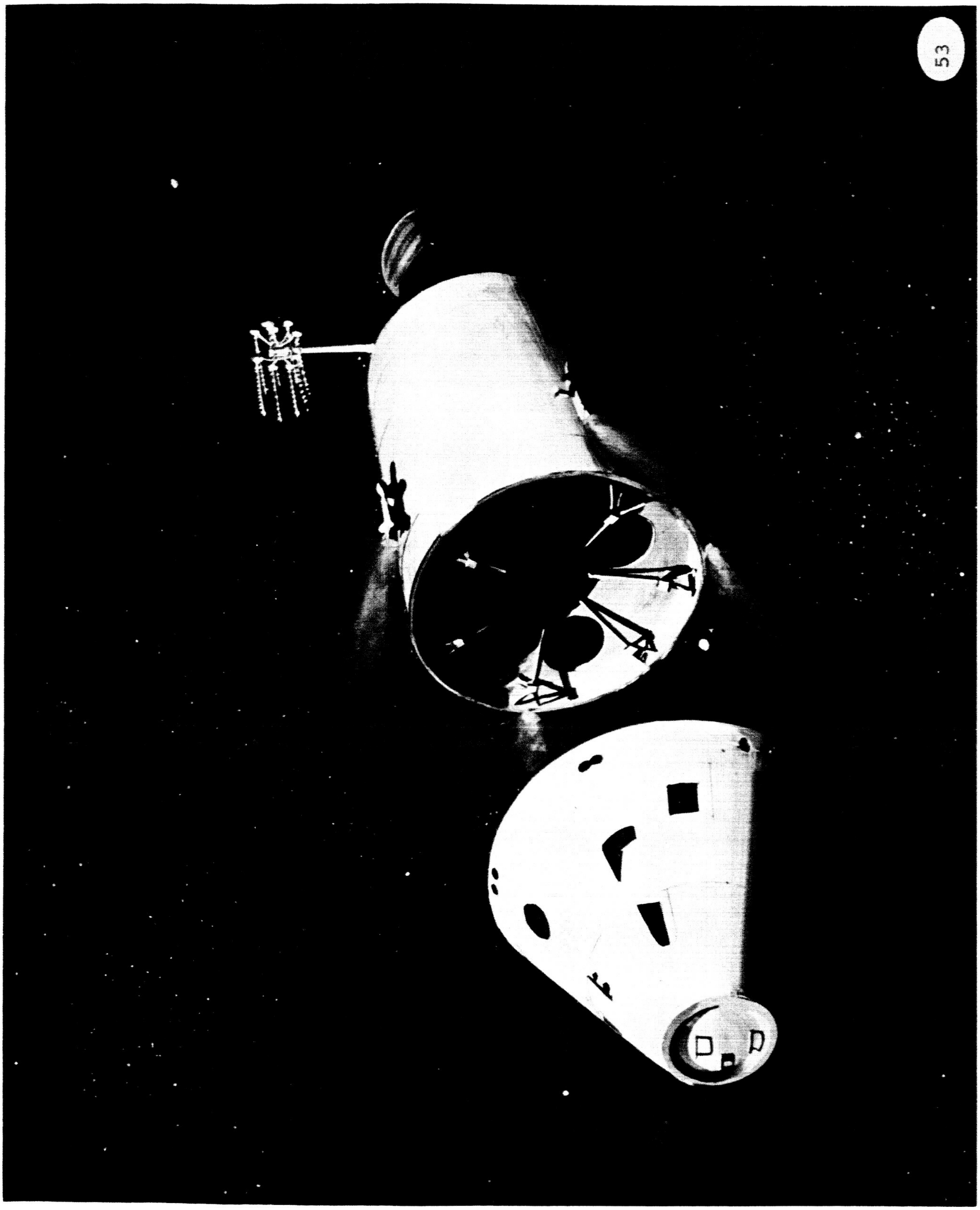

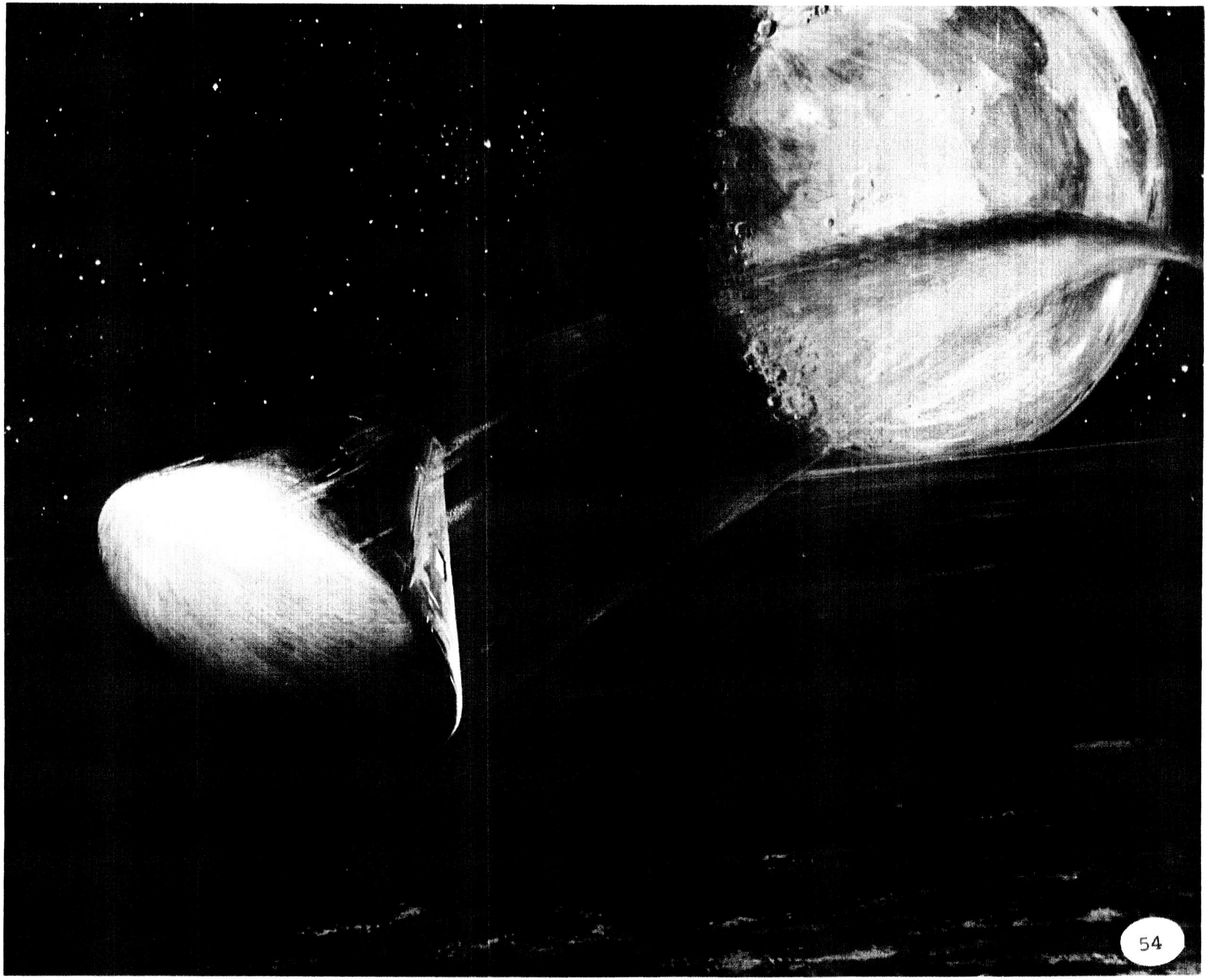

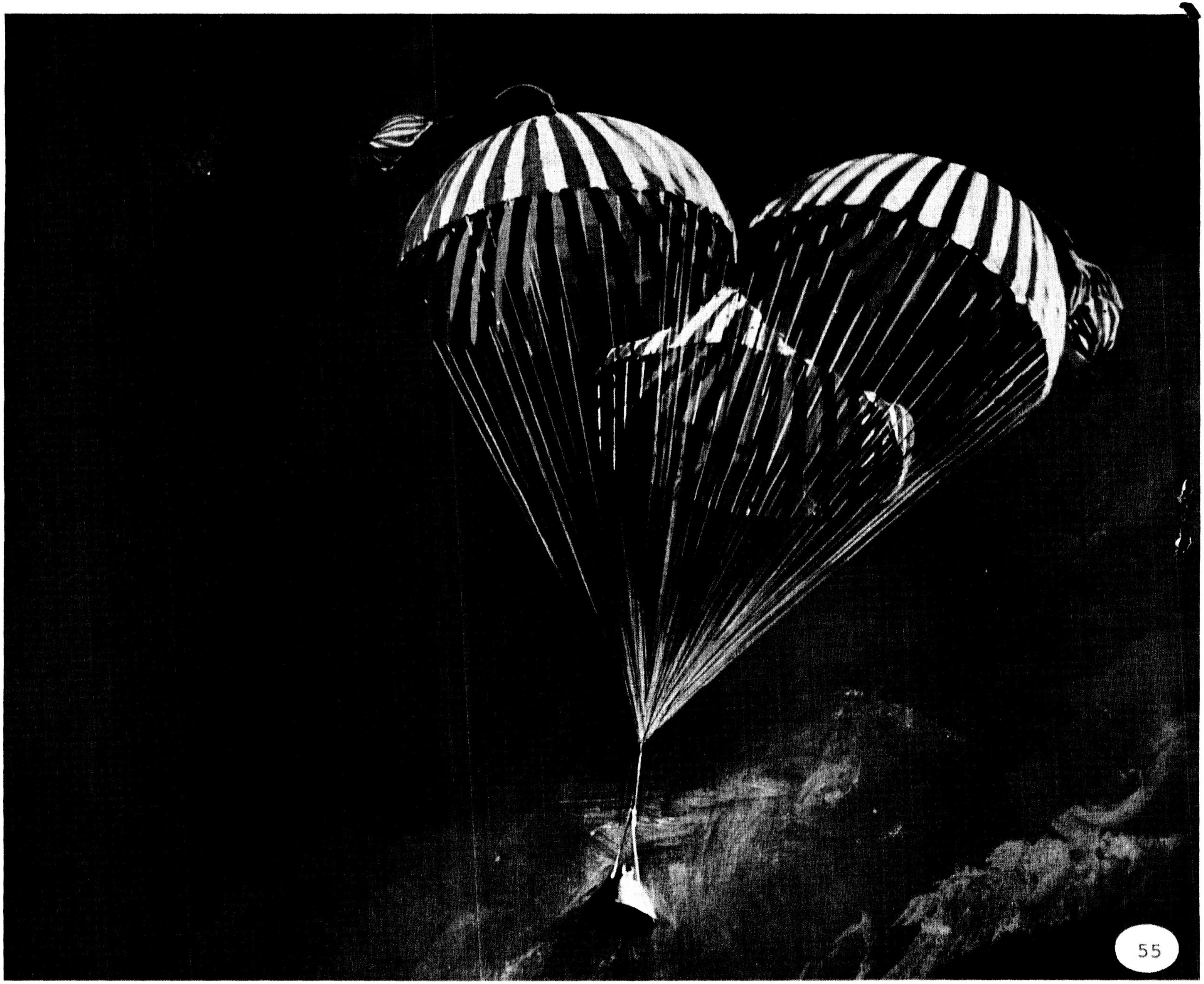
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